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# Solar Dynamic Power System Definition Study

## Final Report

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# FOREWORD

This report summarizes the work accomplished under NASA Contract NAS3-24864, "Solar Dynamic Power System Definition Study", sponsored by the NASA Lewis Research Center. The work was performed by the Rocketdyne Division of Rockwell International Corporation over the period of November 1985 through September 1987. The Principal Investigator was Wayne E. Wallin. Jerry M. Friefeld was the Program Manager.

The program was comprised of two related parts:

1. A system definition study of the application of solar dynamic power systems to NASA, civil, and military missions other than the Phase I configuration Space Station, and
2. An assessment of the survivability and an evaluation of the hardening potential of the solar dynamic power systems.

The system definition study compared both different types of dynamic power generating systems and photovoltaic power generating systems. The system definition study conceptual designs were used as initial input to the survivability and hardening studies, which considered natural environmental threats and hostile threats. Detailed results of the survivability and hardening work are classified and are reported under separate cover. (NASA CR 180878: Solar Dynamic Power System Definition Study - Survivability Analysis).

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# ACRONYMS

| | |
|---|---|
| ac | alternating current |
| A/D | advanced development |
| ALIP | annular linear induction pump |
| AMO | air mass zero |
| ASD | Advanced Solar Dynamics |
| ATP | advanced technology program |
| BCDU | battery charge discharge unit |
| BOL | beginning-of-life |
| CBC | closed Brayton cycle |
| COMSAT | communication satellite |
| DBS | direct broadcast satellite |
| dc | direct current |
| DCCU | dc control unit |
| DCSU | dc switch unit |
| DOD | Department of Defense |
| ECC | electronic components cooler |
| ECG | epitaxial crystal growth |
| EM | electromagnetic |
| EOL | end-of-life |
| ESS | Extendible Support Structure |
| f/D | focal length/diameter ratio |
| FC-75 | 3M fluorinated organic liquid |
| FLIP | flat linear induction pump |
| FPSE | free piston Stirling engine |
| FRUSA | Flexible Rolled-Up Solar Array |
| GCR | geometric concentration ratio |
| GEO | geosynchronous earth orbit |
| GTRI | Georgia Tech Research Institute |
| HP | heat pipe |
| HX | heat exchanger |
| ID | inside diameter |
| IGT | Institute of Gas Technology |
| IR&D | independent research and development |
| KEW | kinetic energy weapon |

## ACRONYMS (Continued)

| | |
|---|---|
| LANL | Los Alamos National Laboratory |
| LASA-B | Laser Atmospheric Sounder and Altimeter |
| L/D | length/diameter ratio |
| LEO | low-earth orbit |
| LeRC | Lewis Research Center |
| LIDAR | Laser Identification Defection and Ranging |
| MBSU | main bus switch unit |
| MEO | mid-earth orbit |
| MFI | multi-foil insulation |
| MIU | main inverter unit |
| MPS | material processing in space |
| MSFC | Marshall Space Flight Center |
| MTI | Mechanical Technology Incorporated |
| NaK 78, NaK | electric mixture of sodium and potassium |
| NPB | neutral particle beam |
| NSTS | National Space Transportation System |
| OAST | Office of Aeronautics and Space Technology |
| OD | outside diameter |
| OMV | orbital maneuvering vehicle |
| ORC | organic Rankine cycle |
| ORU | orbital replacement unit |
| OTV | orbit transfer vehicle |
| PCM | phase change material |
| PCU | power conversion unit |
| PDCA | power distribution and control assembly |
| PDCU | power distribution and control unit |
| P/L | pumped loop |
| PLR | parasitic load radiator |
| PMAD | power management and distribution |
| PMC | power management controller |
| POP | polar orbit platform |
| POPs | Program Operational Plans |
| PR | pressure ratio |
| PSCU | power source control unit |

## ACRONYMS (Concluded)

| | |
|---|---|
| PV | photovoltaic |
| PVC | photovoltaic controller |
| RTG | radioisotope thermoelectric generator |
| SAFE | Solar Array Flight Experiment |
| SD | solar dynamic |
| SDI | Strategic Defense Initiative |
| SDPSD | Solar Dynamic Power System Definition Study |
| SOA | state-of-the-art |
| SOW | statement of work |
| SPDE | Space Power Demonstrator Engine |
| SRP | splined radial panel |
| SSE | Space Stirling Engine |
| SSEC | Solar System Exploration Committee |
| SSU | sequential shunt unit |
| STAS | Space Transportation Architecture Study |
| STSD | Rockwell's Space Transportation Systems Division |
| TEM | thermoelectric electromagnetic |
| TES | thermal energy storage |
| UV | ultraviolet |
| VRM | Venus Radar Mapper |
| vs | versus |
| 3M | Minnesota Mining and Manufacturing |

# 1.0 SUMMARY

A design and analysis trade study was conducted for the NASA-LeRC Office of Aeronautics and Space Technology to compare different solar dynamic cycles for application to future space missions beyond the NASA Phase I Space Station. Closed Brayton cycle, alkali-metal Rankine cycle, and free piston Stirling cycle were evaluated and also compared with two photovoltaic power systems (planar silicon cell array and gallium arsenide concentrator array). The analysis and comparison utilized Space Station level of technology where possible, including the Space Station advanced development contract work on the solar concentrator, solar receiver, and heat pipe radiator. The objectives of the study were: to determine the potential for performance improvements (reduced weight and area) for solar dynamic power, to address areas of technical advancement needed to realize the performance improvements, and to recommend an advanced technology program to address those areas.

The study compared both 35 kWe and 7 kWe power levels, each sized to provide 20 kHz ac power to the user. Thermal energy storage material was selected as LiF salt. The systems were compared on the basis of 7-year end-of-life performance in low earth orbit. Comparative results for the 35 kWe power level, based on weight, placed the Stirling cycle about 22% below the Brayton cycle, whereas the silicon array was about 37% higher and the concentrator array about 35% higher than the Brayton cycle. For the 7 kWe power level, the respective values were: -18%, +27%, and +26%. The Stirling cycle also resulted in lowest area as well as lowest power system weight, as a result of the high cycle efficiency expected for Stirling. The alkali-metal Rankine cycle was eliminated from the comparison during the course of the trade study.

Conceptual designs were developed for the Brayton cycle and two Stirling cycle configurations for each power level. A study of hardenability potential for the conceptual designs was performed, indicating that significant improvements can be realized at the expense of some increase in weight. A technology development program was prepared to address areas wherein significant performance improvements could be realized relative to the current state-of-the-art as represented by Space Station.

## 2.0 EXECUTIVE SUMMARY

A preliminary design and analysis study was conducted to compare three different solar dynamic power cycles for application to missions beyond the Phase I Space Station. The Solar Dynamic Power System Definition (SDPSD) Study contract compared closed Brayton cycle (CBC), Stirling cycle, and alkali-metal Rankine cycle dynamic power systems. Two photovoltaic power systems, a planar silicon array and a gallium arsenide (GaAs) concentrator array, were compared with the dynamic systems. Where possible, all systems were based on Space Station level of technology. The results of the design and analysis study were used to recommend a technology development program to address areas of technical concern.

The technology development program addresses the following key issues:

1. Development of light weight concentrators for different mission environments
2. Higher temperature Brayton and Stirling engine operation
3. Stirling engine test experience
4. Stirling engine code development and correlation
5. Heat pipe Stirling engine and solar receiver integration
6. Thermal energy storage (TES) conductance enhancement
7. Solar receiver materials selection

### 2.1 SCOPE OF WORK

The NASA Space Station has ushered in the need for higher electric power production capability in space. Space Station Phase B trade studies (ref. 1) have clearly established the performance potential and life cycle cost advantages of solar dynamic power systems. The performance advantages are high efficiency, low weight, low drag area, and the potential for long life with minimal degradation. To meet the power needs of future spacecraft, beyond the Phase I Space Station, NASA Lewis Research Center (LeRC) has initiated an Advanced Solar Dynamic (ASD) Project (under an Office of Aeronautics and Space Technology (OAST) program), to develop the technology for the next generation of solar dynamic power systems. One of the first projects for ASD was the

SDPSD study contract, to determine what level of power system performance could be obtained using state-of-the-art (SOA) technology, and to recommend a technology development program to resolve potential technology issues associated with Advanced Solar Dynamic Systems.

The SDPSD program focused on:

1. Defining future NASA, civil, and military space missions of the time period beyond 1992 other than the Space Station Phase I missions*
2. Defining and comparing application of Brayton, Rankine, and Stirling cycles to two selected, representative missions
3. Identification of technology advancement needs
4. Recommendation of an advanced technology program to address those needs

*Note: Phase I Space Station will be 75 kWe, powered by photovoltaic arrays. Phase II will add 50 kWe of solar dynamic power, with scarring to accommodate a further addition of 50 kWe of solar dynamic power.

The Space Station Power System Phase B design data (ref. 1) were used both as a point of departure for design of the SOA higher temperature solar dynamic power systems, and as correlation data to validate the power system analysis codes used to study the Brayton, Rankine, and Stirling power systems. Study results from ongoing Space Station Advanced Development (A/D) programs by Boeing Aerospace Company (L.M. Sedgwick, Boeing, Seattle, WA, Contract NAS3-24669, "Solar Dynamic Heat Receiver Technology"), Grumman Space Systems (ref. 2), and Harris Corporation (ref. 3) were also utilized. The space station photovoltaic power system design changed between the Phase B studies (where the power system was proposed as a hybrid design of photovoltaic and solar dynamic) and the Phase I all-photovoltaic design configuration recently selected for Space Station. The Phase I photovoltaic design data for arrays, batteries, thermal control, and power management and control (PMAD) were used as a point of departure for this study.

The survivability study provided an assessment of the survivability and an evaluation of the hardening potential of the several solar dynamic power system conceptual designs developed during the system definition study; namely, the closed Brayton cycle, Stirling pumped loop receiver, and Stirling heat pipe receiver power cycles. Conceptual designs of hardened solar dynamic system components (i.e., the solar concentrator, the radiator, the power management subsystems, and the receiver/thermal storage/engine package) were developed for each of the system concepts. The results were used to make recommendations on development needs to enable solar dynamic systems to meet the specified natural and hostile environmental threat criteria.

## 2.2 SELECTED MISSIONS

Mission selection entailed review of future missions and selection of two representative missions with different power levels and orbital characteristics. The selected missions were used for study and evaluation of the Brayton, Rankine, and Stirling solar dynamic power cycles.

A large number of future missions were reviewed, as indicated in figure 2.2-1: power levels from 3 to 300 kWe; earth orbital, interplanetary, orbit transfer, and lunar; geocentered orbits from equatorial to polar, and low-earth orbit (LEO) through geosynchronous earth orbit (GEO); and included NASA missions beyond Phase I Space Station, commercial missions, and generic military missions. Factors considered included: power level and electrical characteristics, orbital characteristics, life, and type of mission (astrophysics, communications, earth sciences, lunar, etc.).

The representative missions were to be selected one each from two power categories: 3 to 25 kWe and 25 to 300 kWe. With the concurrence of NASA-LeRC, two missions were synthesized from the mission data base, rather than choosing two specific missions, in order to emphasize technology issues for the power systems. The missions are:

| Power, kWe | Altitude, km | Orbit Inclination | Service |
|---|---|---|---|
| 35 | 500 | 28.5° | Serviceable |
| 7 | 1200 | Variable* | Unserviceable |

*Note: An orbit inclination of 60° was used to size the 7 kWe power systems; this selection results in the condition of no eclipse period for some orbits.

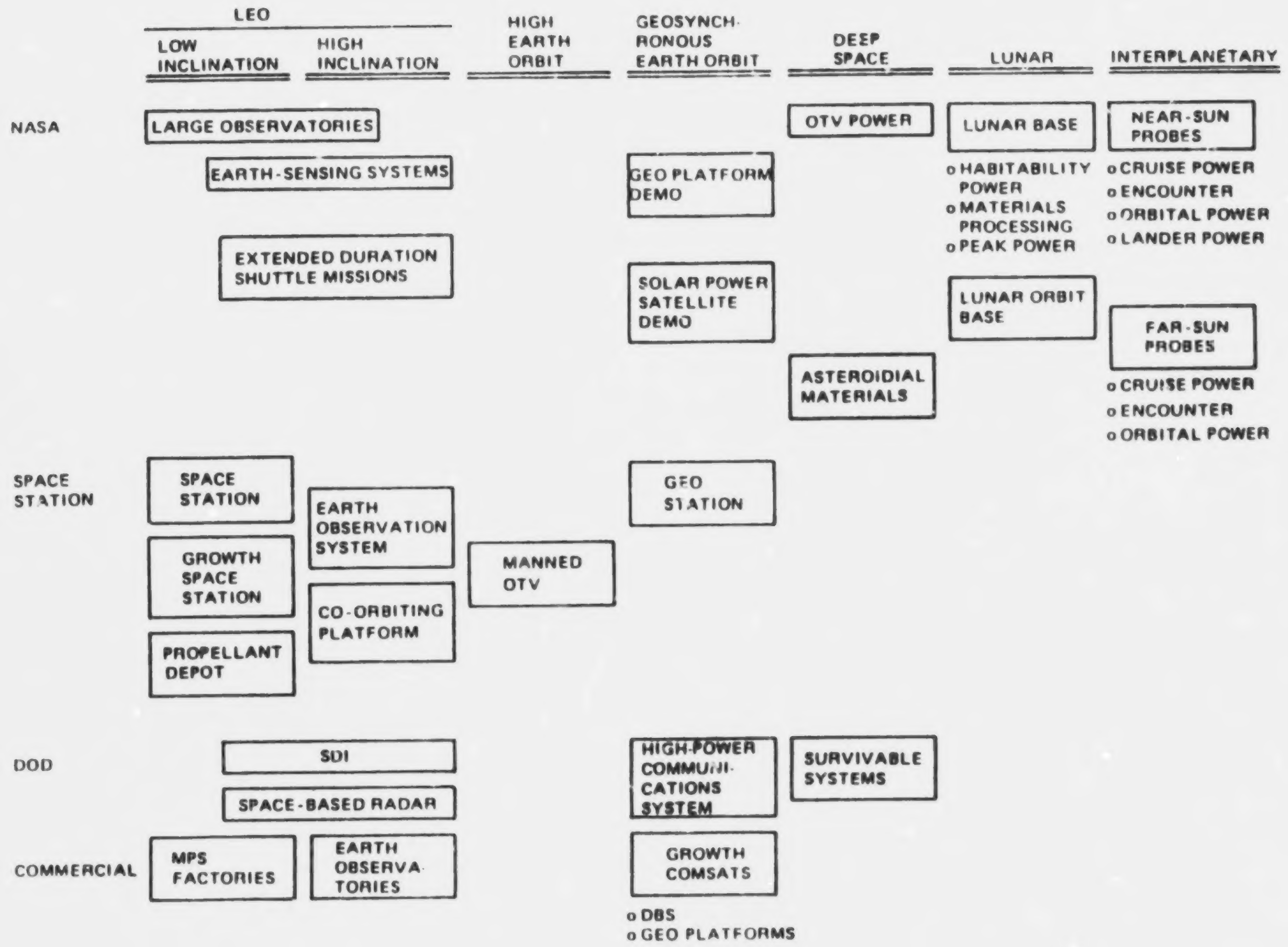

LEO
LOW INCLINATION
HIGH INCLINATION
HIGH EARTH ORBIT
GEOSYNCHRONOUS EARTH ORBIT
DEEP SPACE
LUNAR
INTERPLANETARY
NASA
LARGE OBSERVATORIES
EARTH-SENSING SYSTEMS
EXTENDED DURATION SHUTTLE MISSIONS
GEO PLATFORM DEMO
SOLAR POWER SATELLITE DEMO
OTV POWER
ASTEROIDIAL MATERIALS
LUNAR BASE
o HABITABILITY POWER
o MATERIALS PROCESSING
o PEAK POWER
LUNAR ORBIT BASE
NEAR-SUN PROBES
o CRUISE POWER
o ENCOUNTER
o ORBITAL POWER
o LANDER POWER
FAR-SUN PROBES
o CRUISE POWER
o ENCOUNTER
o ORBITAL POWER
SPACE STATION
SPACE STATION
GROWTH SPACE STATION
PROPELLANT DEPOT
EARTH OBSERVATION SYSTEM
CO-ORBITING PLATFORM
MANNED OTV
GEO STATION
DOD
SDI
SPACE-BASED RADAR
HIGH-POWER COMMUNICATIONS SYSTEM
SURVIVABLE SYSTEMS
COMMERCIAL
MPS FACTORIES
EARTH OBSERVATORIES
GROWTH COMSATS
o DBS
o GEO PLATFORMS

Figure 2.2 1. Future Space Missions

## 2.3 CHARACTERIZATION OF SOLAR DYNAMIC POWER CYCLES

The characterization of the power cycles utilized in-house developed solar dynamic power system codes to compare the Brayton, Rankine, and Stirling cycle systems analytically. Each cycle was evaluated at each of the two power levels. Two types of photovoltaic power systems utilizing planar silicon arrays and GaAs concentrator arrays were also evaluated for comparison to the solar dynamic power systems.

### 2.3.1 Results of the Solar Dynamic and Photovoltaic Power Systems Comparison

Results of the design and analysis study for the two selected missions are presented in figures 2.3.1-1 and 2.3.1-2. Each figure compares weight and area of five power systems: Brayton dynamic cycle, two Stirling dynamic cycles (heat pipe receiver configuration and pumped loop receiver configuration), silicon photovoltaic, and GaAs photovoltaic. Results of the Rankine cycle are not included as the cycle was not found to be competitive with the other dynamic cycles due to lower cycle efficiency and excessive power system weight. Solar dynamic trade study results are presented in section 6.4 and design descriptions are presented in section 7.2. Photovoltaic power system design information is presented in section 9.

Deployed area, weight, and launch volume are important system parameters when comparing alternate power system concepts. Weight primarily effects transportation cost to orbit. Deployed area perpendicular to motion primarily effects drag for lower altitude orbits and the cost of drag makeup propellant for altitude reboost. Area also effects the probability of micrometeoroid and debris impact. Launch volume rather than launch weight will be the limiting factor in some instances, thereby becoming the primary effect on transportation cost to orbit. Weight and area, being direct outputs of the trade study, were chosen for initial comparison of the power system designs. The areas of the solar dynamic power systems are presented as the sum of the concentrator gross aperture area plus radiator sail (planform) area. Equivalent drag area is less than the sum of the areas due to the orientation of each component with respect to the orbital velocity vector.

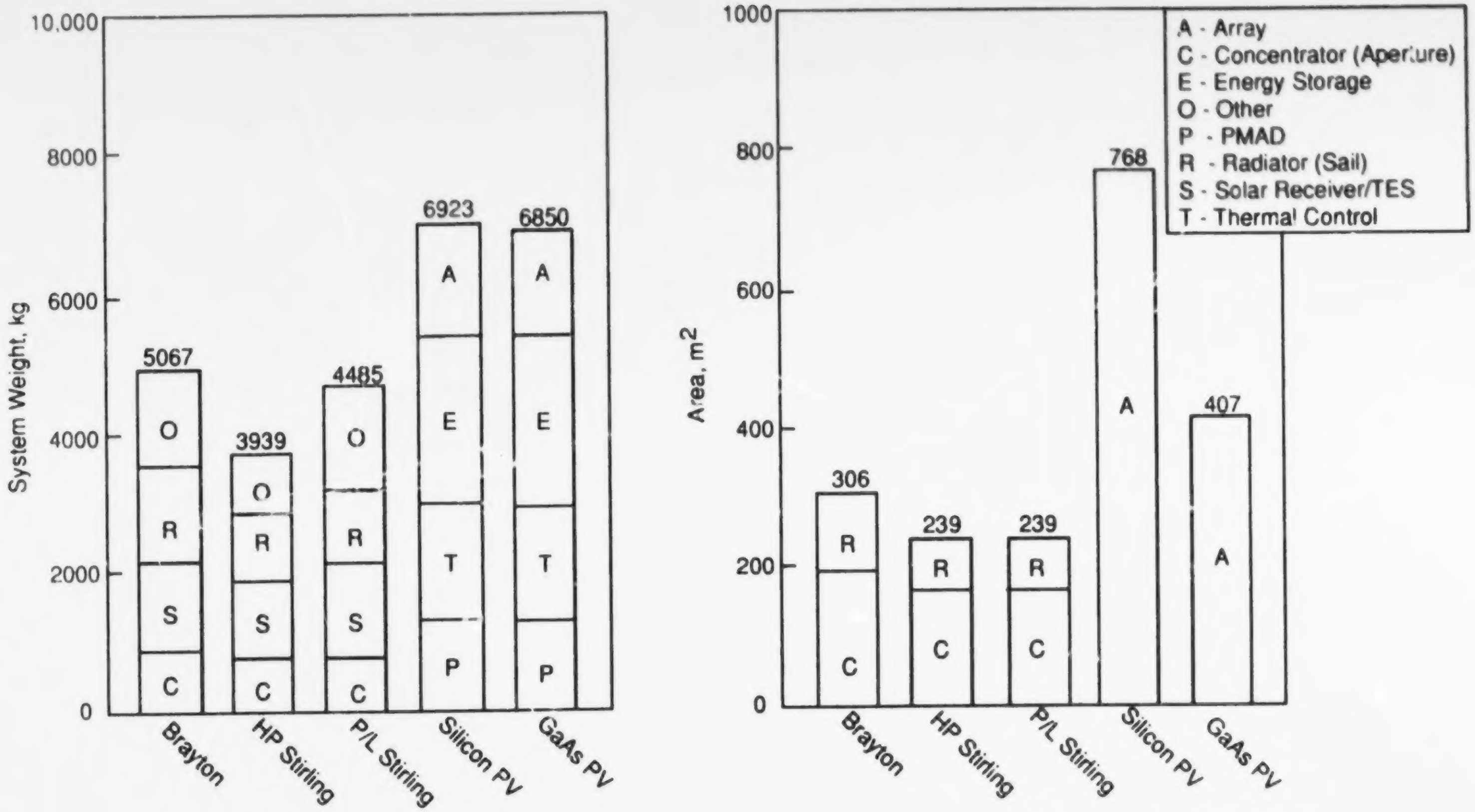


Figure 2.3.1-1. 35 kWe Solar Power Systems - System Weight and Area

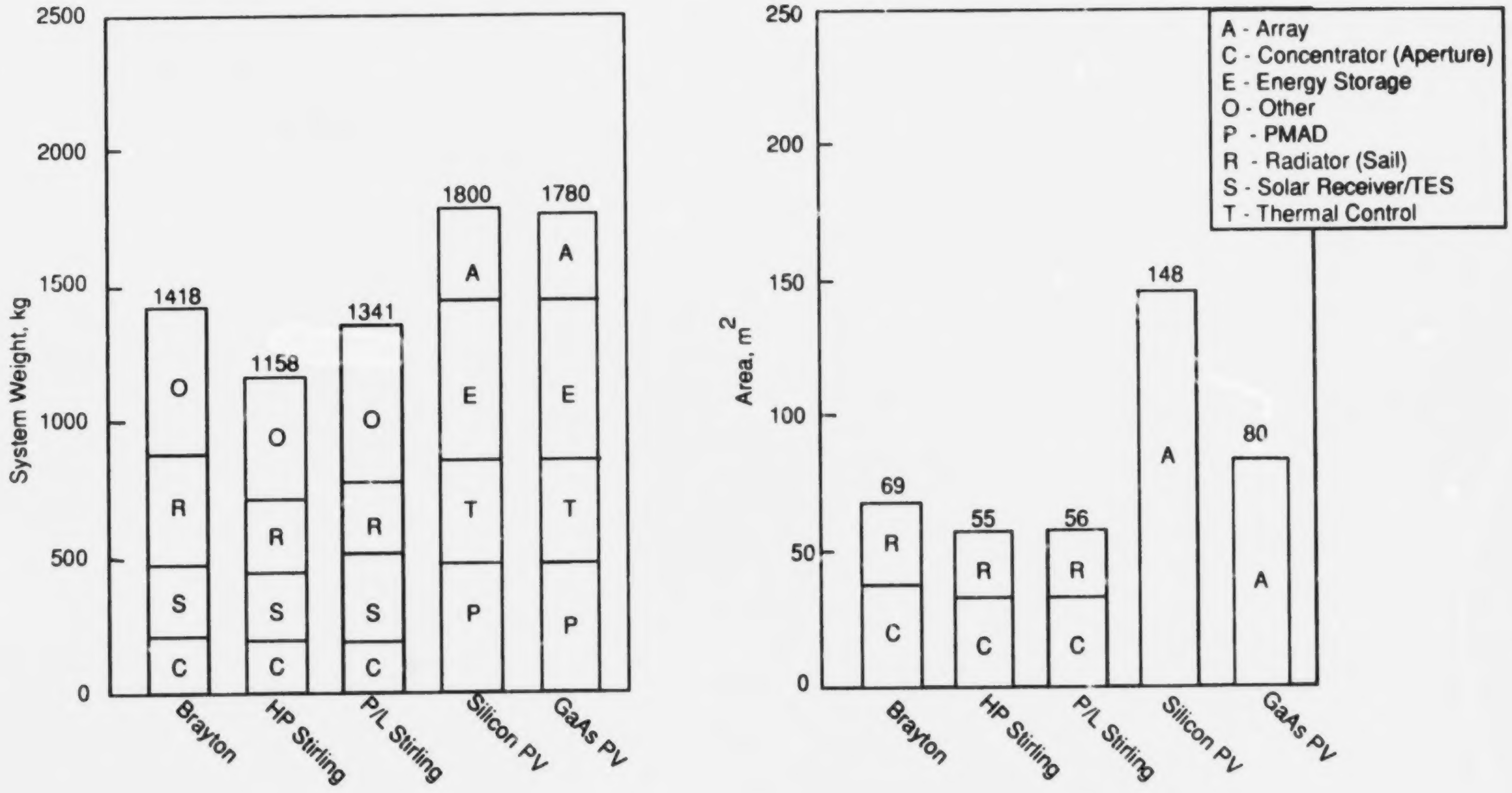


Figure 2.3.1-2. 7 kWe Solar Power Systems - System Weight and Area

Figures 2.3.1-1 and 2.3.1-2 show a performance (weight and area) advantage of the solar dynamic power systems over the photovoltaic systems, and show that Stirling cycle performance is superior to the Brayton cycle. The study examined two Stirling cycle solar receiver design concepts, with the heat pipe design resulting in reduced weight and area as compared to the pumped loop design. Additional information for the Brayton cycle and heat pipe Stirling cycle is summarized in table 2.3.1-1. State point diagrams for the 35 kWe Brayton cycle and the 35 kWe heat pipe Stirling cycle are presented in figures 2.3.1-3 and 2.3.1-4. Additional tabular data and state point diagrams for all six solar dynamic cycle designs may be found in section 6.4.1.

### 2.3.2 State-of-the-Art

The study was based on near-term SOA for the concentrator, receiver, radiator, and power conversion technologies to establish a technology base. Each of the power systems has evolved to a different level of maturity and confidence as to SOA performance, weight, etc. For example, open-cycle and closed-cycle Brayton experience is more extensive than either the free piston Stirling cycle or the alkali-metal Rankine cycle, and experience with planar silicon arrays is more extensive than with GaAs concentrator arrays. The near-term SOA data base sources used for the design and analysis studies of the solar dynamic cycles are indicated in table 2.3.2-1.

The power system computerized design codes used common algorithms for description of the concentrator, receiver, and radiator. Since the focus of the study was comparison of the power conversion cycles rather than other power subsystems, various concentrator and radiator configurations were not examined, as the effect of such subsystem trades would tend to have had a similar effect upon the different power cycles.

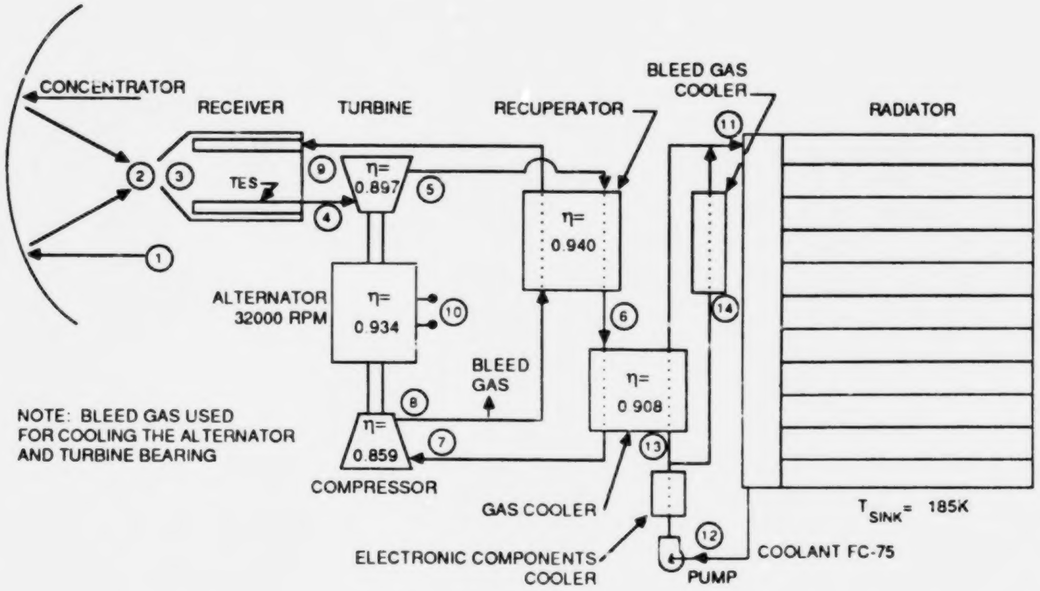
The truss hex concentrator was chosen as SOA because it is the baseline for the Space Station, although it was recognized that lighter weight concepts are under development which may prove to be more appropriate for more weight-sensitive missions such as geosynchronous earth orbits (GEO). The Grumman dual-slot heat pipe was chosen as the SOA radiator concept appropriate to an unserviceable spacecraft configuration, whereas, a lighter weight pumped

Table 2.3.1-1. Solar Dynamic Power Systems Design Data Summary

| Parameter | 35 kWe | | 7 kWe | |
|---|---|---|---|---|
| | Brayton | Stirling(1) | Brayton | Stirling(1) |
| Concentrator gross aperture area, $m^2$ (2) | 196 | 168 | 37.4 | 30.4 |
| Radiator radiant area, $m^2$ (3) | 211 | 137 | 58.8 | 40.8 |
| Radiator sail area, $m^2$ | 110 | 71 | 31.7 | 22.2 |
| Solar multiple (4) | 1.607 | 1.607 | 1.467 | 1.467 |
| Excess energy ratio(5) | 1.22 | 1.22 | 1.57 | 1.57 |
| PCU plus alternator efficiency | 0.356 | 0.420 | 0.342 | 0.408 |
| Efficiency - solar to net power | 0.217 | 0.253 | 0.207 | .235 |
| Concentrator, kg | 845 | 742 | 196 | 180 |
| Receiver/TES, kg | 1255 | 1075 | 281 | 254 |
| PCU, alternator, control (PLR), and structure, kg | 878 | 574 | 266 | 158 |
| PCU radiator and electronic cooling radiator, kg | 1471 | 1006 | 421 | 309 |
| Pumps, accumulators, piping and fluid alowance, kg(6) | 78 | 74 | 18 | 40 |
| Power conversion to 20 kHz, kg | 200 | 200 | 134 | 134 |
| Interface structure, kg(7) | 340 | 268 | 102 | 83 |
| Power system weight, kg | 5067 | 3939 | 1418 | 1158 |

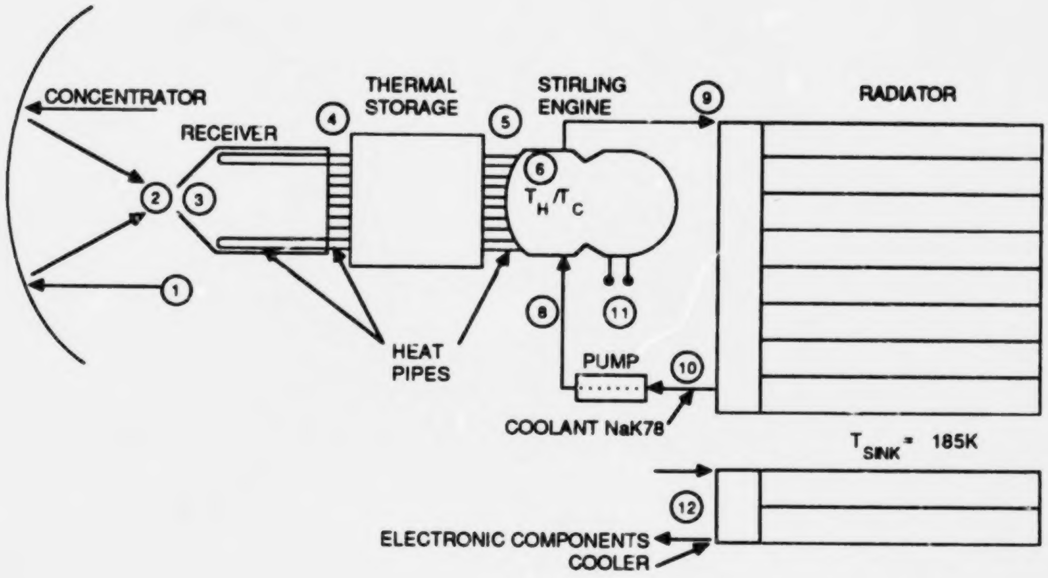
Notes:

1. Heat pipe Stirling configuration
2. Includes blockage and shadow area, and hex segment packing factor (reflective facet area + hex area)
3. Brayton cycle PCU waste heat and electronic cooling loads are combined and serviced by a single radiator. Stirling cycle PCU waste heat load and electronic cooling load are serviced by separate radiators due to temperature differences. Areas include approximately 15% redundancy for seven year lifetime.
4. Orbit period + shortest sun interval for the orbit
5. Orbit (maximum solar intensity times longest sun interval) + (minimum solar intensity times shortest sun interval)
6. Primarily required for the PCU waste heat transport loop. The Stirling cycle utilizes NaK liquid metal for heat transport, and requires an electromagnetic pump.
7. Mounting structure for attachment of the various subsystems including beta-joint interface ring


CONCENTRATOR
RECEIVER
TURBINE
RECUPERATOR
BLEED GAS COOLER
RADIATOR
TES
η= 0.897
η= 0.940
ALTERNATOR 32000 RPM
η= 0.934
BLEED GAS
η= 0.908
NOTE: BLEED GAS USED FOR COOLING THE ALTERNATOR AND TURBINE BEARING
η= 0.859
COMPRESSOR
GAS COOLER
ELECTRONIC COMPONENTS COOLER
PUMP
COOLANT FC-75
T SINK = 185K
①
②
③
④
⑤
⑥
⑦
⑧
⑨
⑩
⑪
⑫
⑬
⑭

| State Point | Power,kW | Temperature, K | Pressure, kPa | Flowrate, kg/sec |
|---|---|---|---|---|
| 1 | 259.7 | | | |
| 2 | 201.9 | | | |
| 3 | 198.6 | | | |
| 4 | | 1086 | 288 | 0.923 |
| 5 | | 890 | 164 | 0.923 |
| 6 | | 437 | 162 | 0.923 |
| 7 | | 300 | 160 | 0.947 |
| 8 | | 400 | 301 | 0.947 |
| 9 | | 858 | 299 | 0.923 |
| 10 | 39.4 | | | |
| 11 | | 398 | | 0.624 |
| 12 | | 280 | | 0.624 |
| 13 | | 286 | | 0.548 |
| 14 | | 286 | | 0.076 |
| 4-5 | 95.5 | | | |
| 5-6 | 222.4 | | | |
| 6-7 | 68.5 | | | |
| 7-8 | 50.3 | | | |
| 8-9 | 222.4 | | | |
| 9-4 | 110.5 | | | |
| 11-12 | 74.8 | | | |
| 12-13 | 3.5 | | | |
| 13-11 | 71.3 | | | |
| 14-11 | 2.8 | | | |

Figure 2.3.1-3. 35 kWe Brayton Cycle State Point Diagram


CONCENTRATOR
RECEIVER
THERMAL STORAGE
STIRLING ENGINE
RADIATOR
①
②
③
④
⑤
⑥
$T_H/T_C$
⑧
⑨
⑩
⑪
⑫
HEAT PIPES
PUMP
COOLANT NaK78
$T_{SINK}$ = 185K
ELECTRONIC COMPONENTS COOLER

| State Point | Power, kW | Temperature, K | Flowrate, kg/sec |
|---|---|---|---|
| 1 | 222.3 | | |
| 2 | 172.6 | | |
| 3 | 167.5 | | |
| 4 | 159.1 | | |
| 5 | 94.5 | | |
| 6 | | 1033 | |
| 8 | | 342 | 1.49 |
| 9 | | 382 | 1.49 |
| 10 | | 341 | 1.49 |
| 11 | 39.7 | | |
| 12 | 3.5 | | |
| 8-9 | 54.8 | | |
| 9-10 | 55.4 | | |
| 10-8 | 0.6 | | |

Figure 2.3.1-4. 35 kWe Heat Pipe Stirling Cycle State Point Diagram

Table 2.3.2-1. Data Bases for Design and Analysis Studies

<table>
<tr><th></th><th colspan="4">REFERENCES</th></tr>
<tr><th></th><th>BRAYTON</th><th>RANKINE</th><th>PUMPED-LOOP STIRLING</th><th>HEAT PIPE STIRLING</th></tr>
<tr><td>CONCENTRATOR</td><td colspan="4">← 1, 3 →</td></tr>
<tr><td>RECEIVER SHELL</td><td colspan="4">← 1 →</td></tr>
<tr><td>RECEIVER INTERNALS AND TES</td><td>Boeing A/D Receiver</td><td>4</td><td>*</td><td>*</td></tr>
<tr><td>POWER CONVERSION UNIT</td><td>1</td><td>4</td><td>5</td><td>5</td></tr>
<tr><td>RADIATOR</td><td colspan="4">← 2 →</td></tr>
<tr><td>FREQUENCY CONVERSION</td><td colspan="4">← 1 →</td></tr>
<tr><td colspan="5">*DESIGN DEVELOPED FOR THIS STUDY</td></tr>
</table>

loop radiator could be more appropriate where servicing is possible as is the case for Space Station. The CBC receiver internals with integral TES were patterned after the Boeing Space Station advanced development design (L.M. Sedgwick, Boeing, Contract NAS3-24669). The liquid-metal pumped loop receiver with remote TES and the heat pipe receiver with TES adjacent to the receiver cavity are design concepts evolved for this study. All receiver designs were scaled for effects of temperature and thermal power. The free piston Stirling engine designs were patterned after the NASA-LeRC sponsored 25 kWe Space Stirling Engine (SSE). The alkali-metal Rankine receiver and engine designs were patterned after similar work performed during the early 1960s. Power output from each of the systems, both solar dynamic and photovoltaic, was subject to frequency conversion to 20 kHz.

Trade studies considered power system weight and area, normally optimizing on the basis of minimum weight. However, realistic design optimization criteria must ultimately be dependent upon mission application considering parameters beyond weight and area, such as: launch volume, SOA readiness, reliability, life-time degradation, and life cycle cost.

### 2.3.3 Peak Cycle Operating Temperatures

The study of peak cycle operating temperatures began with selection of TES salts with high heat of fusion at temperatures above 1090K (1500F). The four candidates are shown in figure 2.3.3-1 along with the $LiF-CaF_2$ eutectic mixture selected for the Space Station CBC cycle. A fifth material was included initially, a eutectic mixture of Mg and Si thought to have a heat of fusion of 1212 kJ/kg. Subsequently, the material was dropped from SOA consideration due to the possibility of much lower heat of fusion and a lack of information on containment compatibility. Heat of fusion for $Mg_2Si$ is indicated over a range, the lower value from reference 4 and the higher value being recent preliminary results (R.P. Wichner, 1988, Oak Ridge National Laboratory, private communication).

Assumptions were made on a variety of power system parameters (concentrator accuracies, receiver losses, temperature ratios and pressure ratios, etc.) in order to carry out the process of system sizing and weight optimization for each power cycle. Many of these parameters were altered and optimized subsequent to the TES optimization study, which is illustrated in the following three figures. Figure 2.3.3-2 shows LiF resulting in a 3.1% lower CBC power system weight than for NaF TES, whereas figures 2.3.3-3 and 2.3.3-4 show LiF resulting in essentially the same Stirling power system weight as for NaF TES. These comparisons did not consider any high-temperature material weight change as might result from switching from superalloys for the CBC and Stirling LiF TES power systems to the use of refractory metals for the higher temperature TES materials. Results of the Space Station A/D receiver work (L.M. Sedgwick, Boeing, Contract NAS3-24669) regarding material strength (creep-rupture) raises questions as to the use of superalloys in receivers with temperatures associated with LiF TES. Appropriate changes in concentration ratio and receiver intercept factor and thermal losses were considered for the different TES materials. A major factor which compounds the differences in TES properties (heat of fusion, density) is the fact that the TES containment weight plus the conductance enhancement weight may be 2-4 times the actual TES material weight itself. In turn, the system weight comparisons can be effected somewhat by the design approach and ingenuity employed in the TES design. The study results of minimum power system weights associated with LiF TES resulted

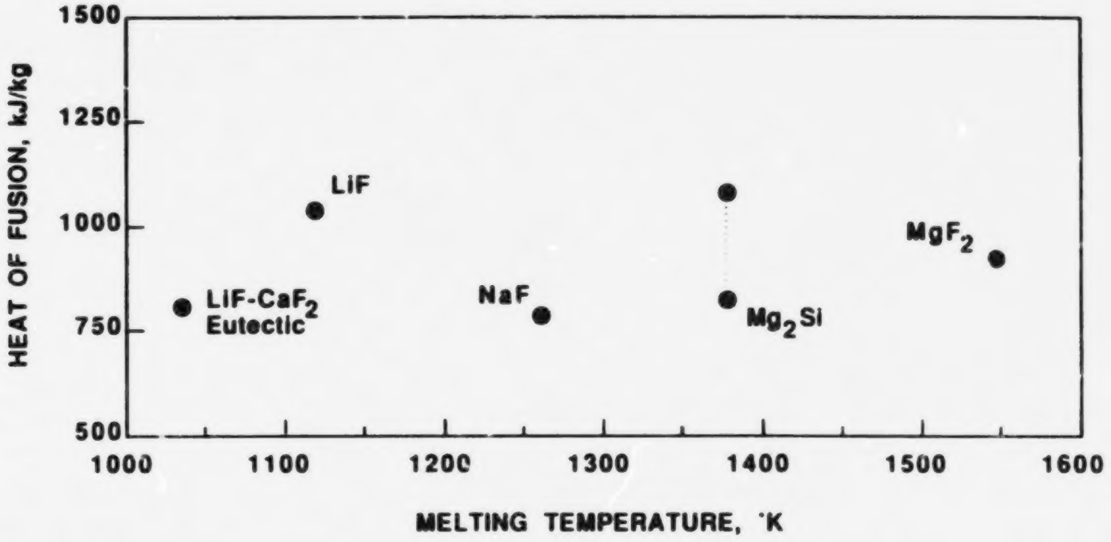


Figure 2.3.3-1. Candidate Thermal Energy Storage Materials

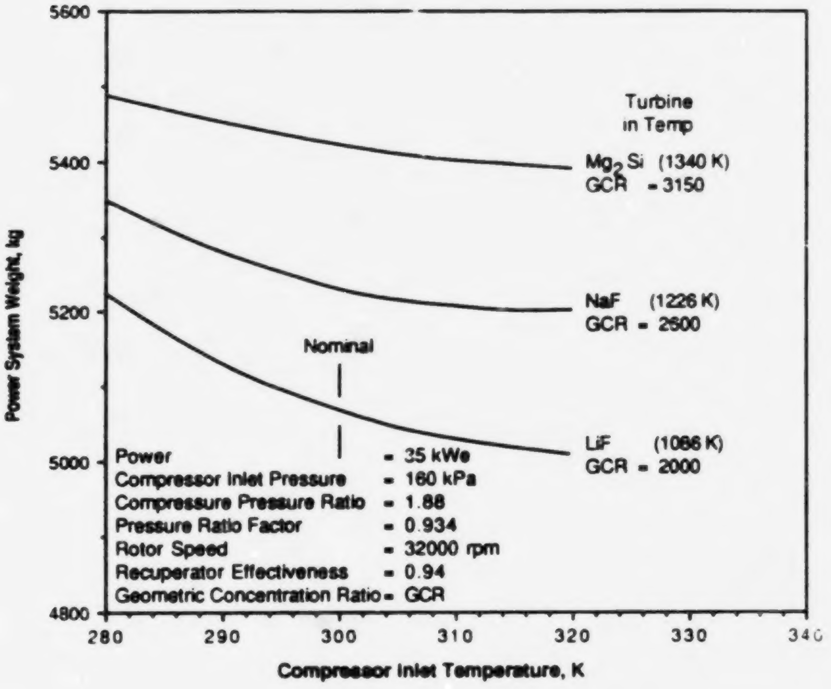


Figure 2.3.3-2. Brayton System Weight vs Compressor Inlet Temperature

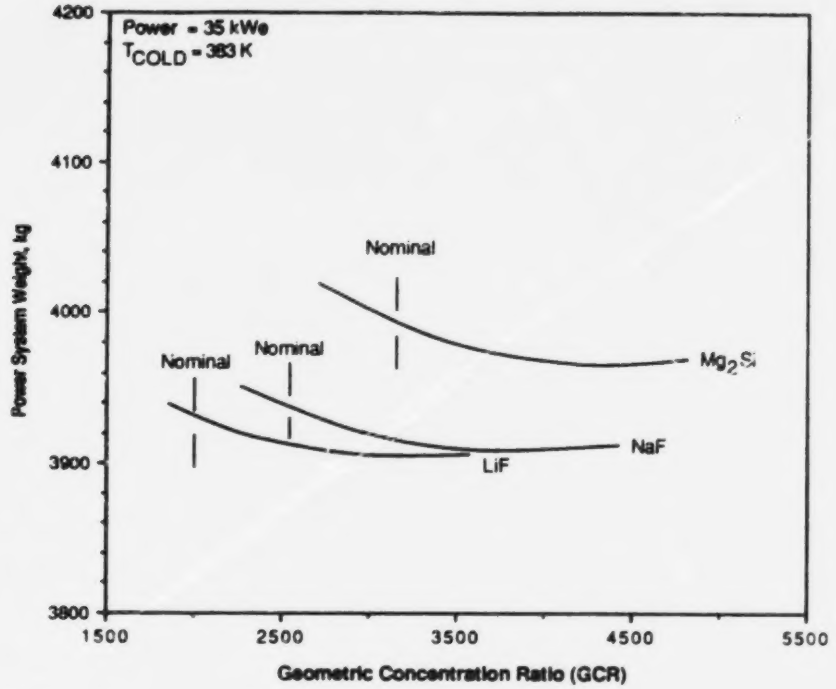


Figure 2.3.3-3. Stirling System Weight vs Geometric Concentration Ratio

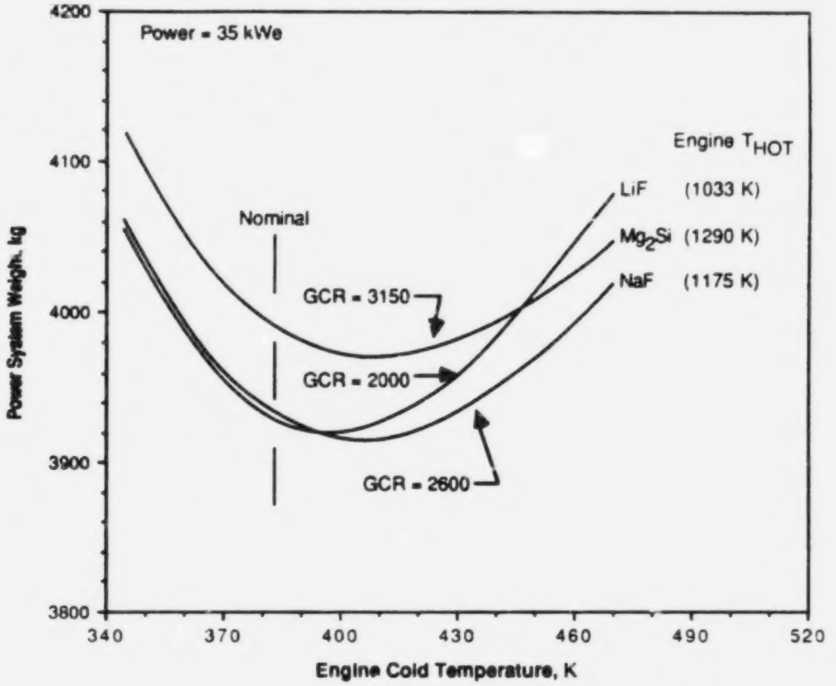


Figure 2.3.3-4. Stirling System Weight vs Engine Cold Te perature

in the selection of LiF for both the CBC and Stirling power cycles.

### 2.3.4 Rankine Cycle

The third dynamic cycle, the alkali-metal Rankine cycle, was found to be incapable of reasonably achieving the high thermodynamic efficiencies realized for CBC (38.1%) or for Stirling (45.6%). Including alternator efficiencies, the CBC and Stirling cycles result in 35.6% and 42.0% power conversion efficiencies, respectively. A single-stage potassium Rankine cycle will result in 15% - 20% efficiency for the range of temperatures considered in this study. Addition of one or two turbine reheats raises thermodynamic efficiency to 25% - 30%, but at the expense of a two- or three-stage turbine and a solar receiver design capable of providing the needed low-pressure reheats. A single-reheat case with 25.2% efficiency using rubidium is illustrated in figure 2.3.4-1, taken from the 15 kWe ASTEC work by Sundstrand, (ref. 4). Considering the Rankine cycle from the standpoint of SOA, it was not found reasonable to pursue even higher temperatures or to examine the use of a number of turbine reheats as a means to further increase thermodynamic efficiency.

Design and analyses for the Rankine cycle were not carried through to the same detail as the CBC or the Stirling cycles because of: the low efficiencies resulting in oversized concentrator, receiver, and radiator; the multi-reheat turbine and receiver complexity; and the peak operating temperatures being much higher than those for LiF. The alkali-metal Rankine cycle was therefore eliminated from further consideration.

### 2.3.5 Power Conversion and Design Margins

The study provided for conversion of the power produced by each of the different dynamic and photovoltaic power systems to the conditions of either 208 or 440 V ac, 20 kHz, single-phase net power output, for both the 35 kWe and 7 kWe designs. Electric power frequency conversion efficiency of 93% was used for the dynamic systems, whereas, an efficiency of 91% was used for the inversion of dc power output from the photovoltaic power systems. These efficiencies are reasonably attainable values, based on analyses subsequent to the Space Station Phase B studies (R.L. Phillips, January 1988, Rocketdyne,

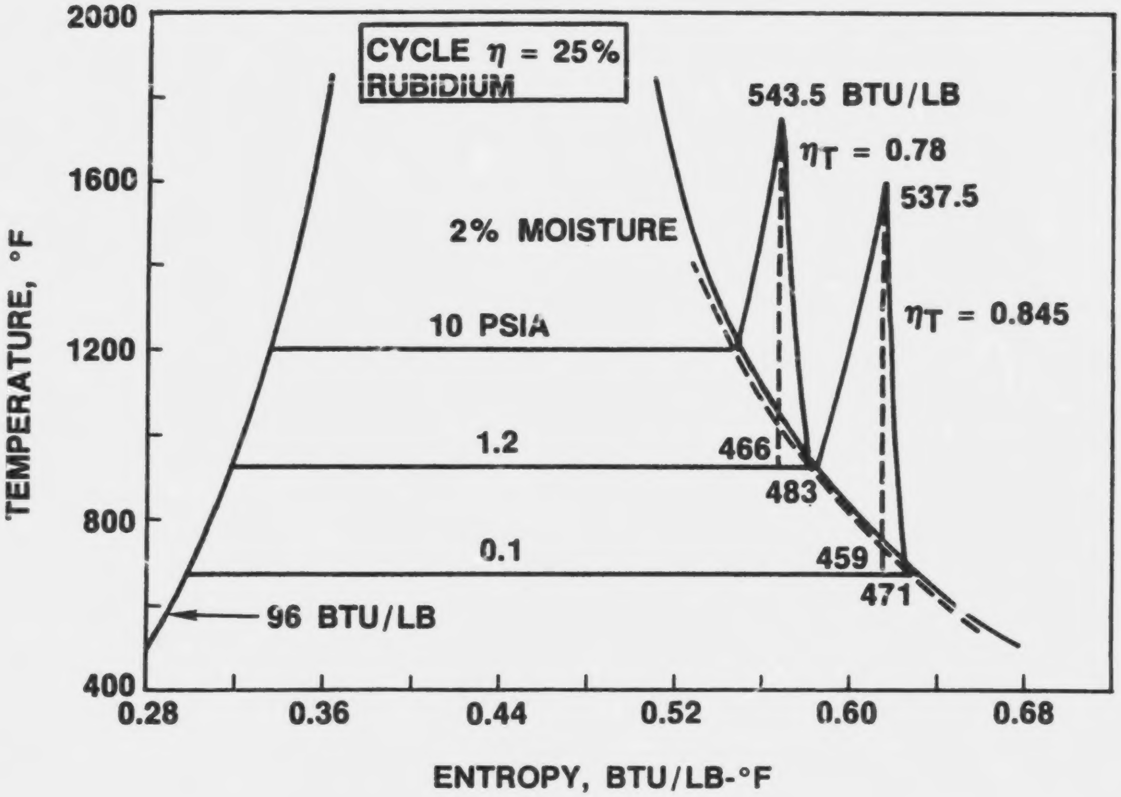


Figure 2.3.4-1. Rankine Thermodynamic Cycle With Single Reheat

Canoga Park, CA, private communication). Correction of alternator output power factor for the Stirling cycle was assumed to be included in the frequency conversion.

A parasitic load radiator (PLR) sized to dissipate the complete alternator power output (allows up to 100% load shedding, for the case of reduced user load or for complete loss of load) was included in each dynamic power system design. This electrical resistance heater is a part of the Space Station engine power control scheme and consumes a minimum of 1.5% to 2.0% of gross output power. Additional parasitic power is required for engine control and concentrator pointing (estimated as 2.5% at 35 kWe and 6% at 7 kWe).

All power systems were designed for end-of-life conditions of 7-10 years, accounting for degradation of photovoltaic cells (7 years) and for the degradation of concentrator and radiator surfaces. The radiator heat pipes were also designed for micrometeoroid hazards in a tradeoff of number of additional panels versus pipe wall thickness, resulting in approximately 15% excess radiator area for redundancy. Dynamic power system weights also include an assumed excess TES (2.5% for the heat pipe receiver, 4% for the pumped loop receiver, and 5% for the other receiver designs) for eclipse operation.

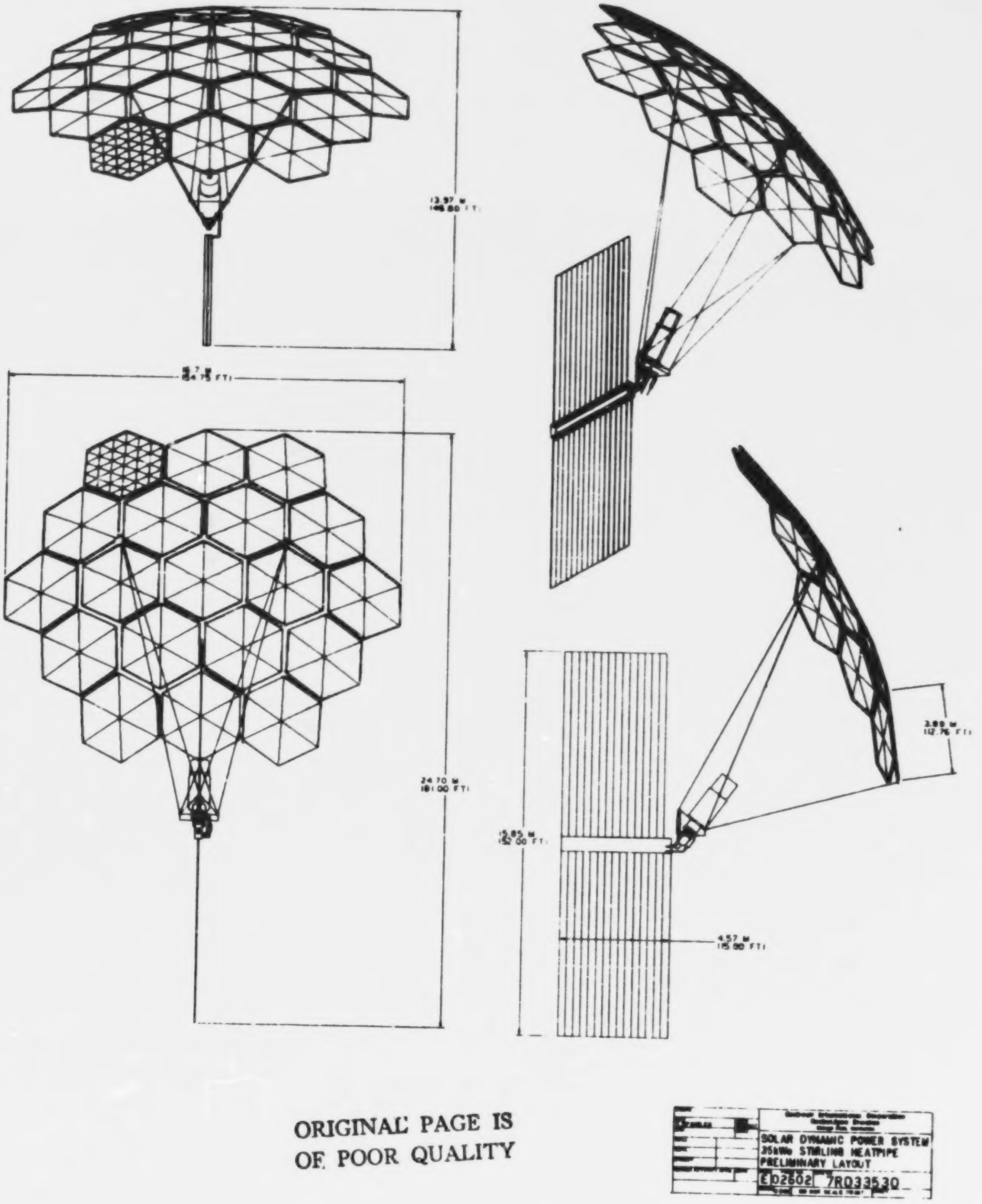
## 2.4 CONCEPTUAL DESIGNS OF THE SOLAR DYNAMIC POWER SYSTEMS

A total of six conceptual designs were developed; two Stirling cycle configurations, and one Brayton cycle configuration, at each of the two power levels, 35 kWe and 7 kWe. As an example, the 35 kWe heat pipe Stirling design is shown in figures 2.4-1 and 2.4-2. The complete set of conceptual designs is presented in section 7.

The solar dynamic power system configurations, each designed for 1121K (1558F) LiF TES, are as follows:

1. A closed Brayton cycle quite similar to that proposed by Garrett Corporation for the Space Station power system, although operating at approximately 1086K (1495F) turbine inlet temperature, rather than 1006K (1350F)
2. A free piston Stirling engine cycle with a pumped liquid metal loop connecting the solar receiver, separate TES unit, and engine
3. A free piston Stirling engine cycle with the thermal connection between the solar receiver and the engine provided by heat pipes, which incorporate integral TES

Some differences, of course, exist between the conceptual designs for this study and the designs resulting from the Space Station Phase B work. A current design configuration for the Space Station CBC solar dynamic subsystem is shown in figure 2.4-3.


13.97 M
(45.80 FT)
24.70 M
(81.00 FT)
3.89 M
(12.75 FT)
15.85 M
(52.00 FT)
4.57 M
(15.00 FT)
SOLAR DYNAMIC POWER SYSTEM
35kWe STIRLING HEATPIPE
PRELIMINARY LAYOUT
E 02602
7R033530

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OF POOR QUALITY

Figure 2.4-1. 35 kWe Heat Pipe Stirling Engine System - Conceptual Design

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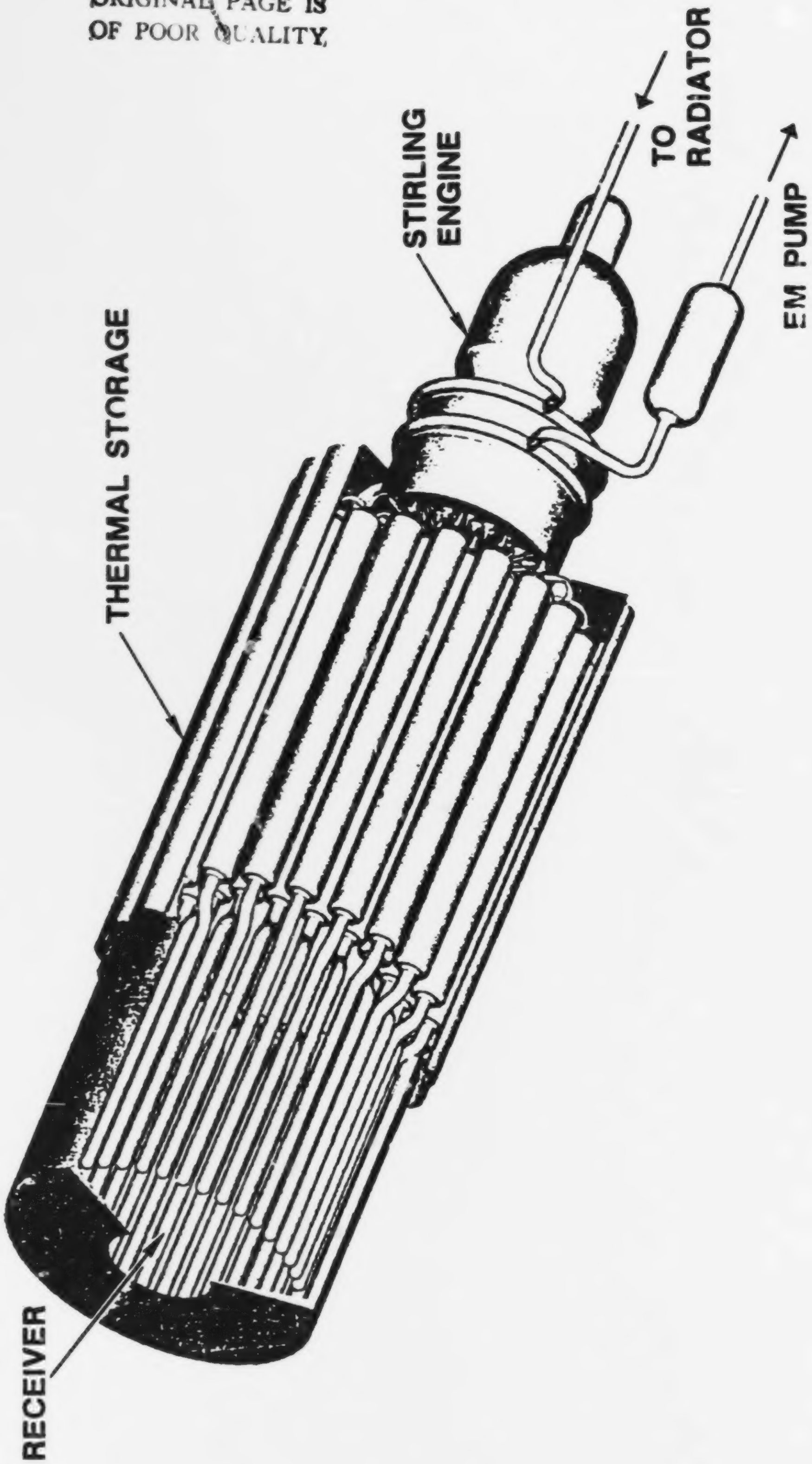
RECEIVER
THERMAL STORAGE
STIRLING ENGINE
TO RADIATOR
EM PUMP

Figure 2.4-2. 35 kWe Heat Pipe Stirling Engine/Solar Receiver

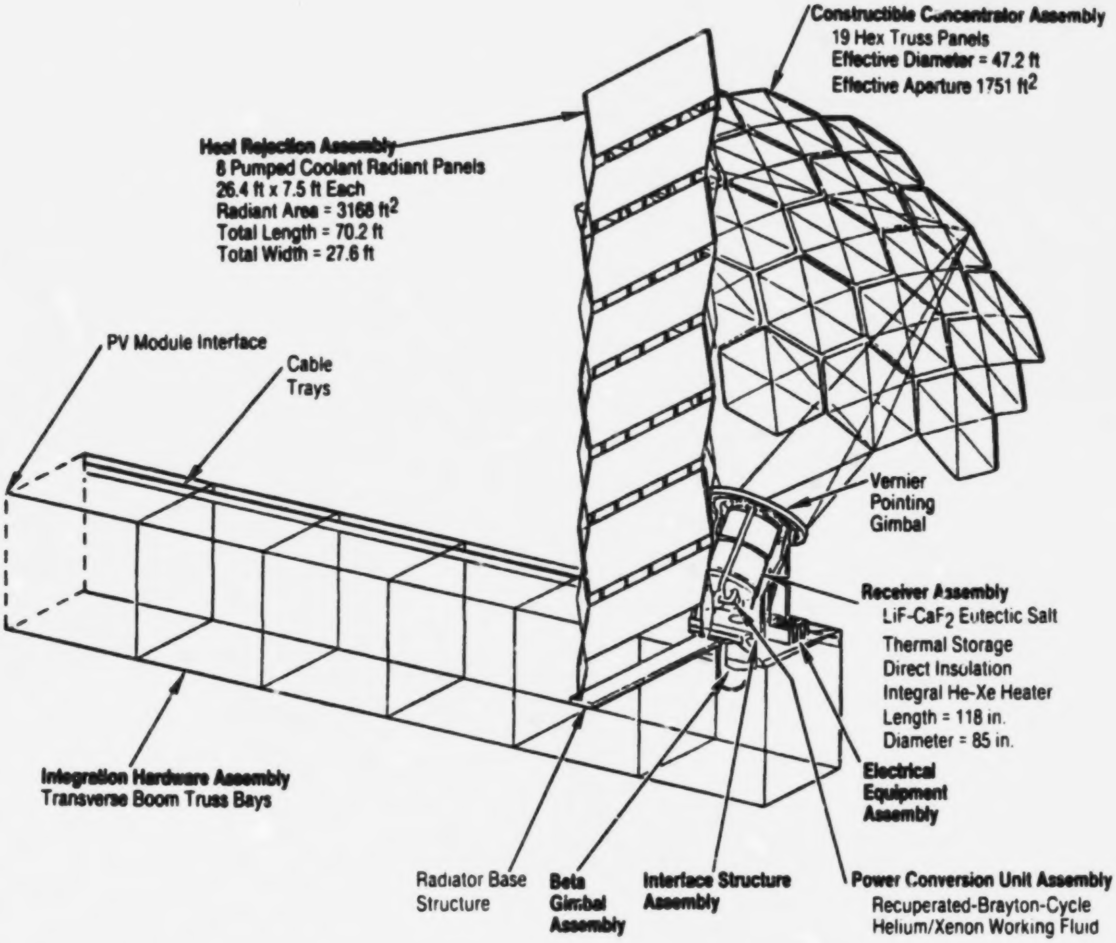
Constructible Concentrator Assembly
19 Hex Truss Panels
Effective Diameter = 47.2 ft
Effective Aperture 1751 ft2
Heat Rejection Assembly
8 Pumped Coolant Radiant Panels
26.4 ft x 7.5 ft Each
Radiant Area = 3168 ft2
Total Length = 70.2 ft
Total Width = 27.6 ft
PV Module Interface
Cable Trays
Vernier Pointing Gimbal
Receiver Assembly
LiF-CaF2 Eutectic Salt
Thermal Storage
Direct Insulation
Integral He-Xe Heater
Length = 118 in.
Diameter = 85 in.
Electrical Equipment Assembly
Integration Hardware Assembly
Transverse Boom Truss Bays
Radiator Base Structure
Beta Gimbal Assembly
Interface Structure Assembly
Power Conversion Unit Assembly
Recuperated-Brayton-Cycle
Helium/Xenon Working Fluid

Figure 2.4-3. 25 kWe Space Station CBC Power Module Design Configuration

A comparison of the Stirling cycle design shown in figure 2.4-1 to that of the Space Station CBC design indicates:

1. A similar concentrator configuration, although the higher concentration ratio (~2000 versus ~1100) will probably require smaller reflective facets for the higher cycle temperature
2. The receiver/TES and power conversion unit (PCU) assemblies are small in proportion to the concentrator and radiator
3. A heat pipe radiator is shown rather than the pumped loop radiator shown for CBC, although that choice is mission dependent in that the pumped loop configuration is not appropriate for a 7-10 year life, unserviceable power system

4. The concentrator two-axis vernier pointing gimbal, beta gimbal assembly, and transverse boom truss with cable trays shown for CBC are not included in the designs for this study

A number of components necessary for a complete power system, beyond the basic concentrator, receiver, engine, and radiator, have been included in the conceptual designs. The following items comprise over 25% of total power system weight:

1. Computers and controls
2. PCU controller and parasitic load radiator
3. Electric power frequency converter
4. Oversizing of the PCU (particularly the alternator) for excess power management (excess solar input)
5. Pumps, fluids, and accumulators
6. PCU mounting structure
7. Redundant radiator panels (micrometeoroid and debris hazard)
8. Electronic component cooling radiator
9. Interface adaptor and superstructure

Alternative design configurations for the concentrator and radiator were not pursued for this study. Such changes would have had similar effects upon each of the dynamic power system designs.

## 2.5 RANKING OF THE SOLAR DYNAMIC POWER SYSTEMS

The purpose of the ranking was to determine suitability of the different solar dynamic power systems for each of the two selected missions. During the analytical characterization part of the study, power system weight was selected as the optimization criterion in order to perform tradeoffs of various system design parameters. However, there are a number of system evaluation criteria which must be considered besides weight, which are: reliability/safety, technology readiness, performance (including weight and area), operability, life cycle cost, and compatibility.

The ranking was performed by comparing each of the Stirling cycle designs to the Brayton cycle design, and then applying the weighting factors for each of the evaluation criteria. Results of the ranking are presented in section 8. The heat pipe Stirling cycle ranked highest, followed by the pumped loop Stirling cycle and, lastly, the Brayton cycle. This order prevailed for each of the missions. However, the Brayton cycle design is based on a great deal of maturity as evidenced by the extensive Space Station CBC Phase B effort. This is not yet true for the Stirling cycles. At this point; however, the ranking results indicate that further development work is warranted for the Stirling cycles.

## 2.6 ADVANCED TECHNOLOGY PROGRAM FOR THE SOLAR DYNAMIC POWER SYSTEMS

The critical development areas arise primary as areas for which little direct experience as yet exists, areas not developed by terrestrial solar dynamic power system experiences. The identification of critical development areas and advanced concepts produced an extensive list of technical issues, as presented in section 10. Six major improvement areas were selected from the list, and are shown in table 2.6-1. Although only one area directly mentions weight, essentially all items mentioned in table 2.6-1 ultimately impact weight of the power system in some way. And, as was seen from results of the ranking task, weight is indeed a primary discriminator.

The critical areas primarily revolve around the following issues. Development of a light-weight concentrator, with minimum losses and degradation, and having a high concentration ratio. The concentrated sunlight is absorbed as heat by a high temperature receiver, with integrated thermal storage for eclipse being provided by the thawing/freezing of an encapsulated salt or other material. The TES material contracts perhaps 25% upon freezing thus creating a void management problem, has a low conductivity in both the liquid and solid state, and is typically highly corrosive. Some considerable design, analysis, and test effort is underway for the CBC type of receiver/TES. However, the heat pipe Stirling receiver, which through heat pipes, passively supplies heat to/from the TES and to the high-temperature-ratio free piston Stirling engine, is at present no more than a conceptual design. The planned 25 kWe SSE $T_H/T_C = 2.0$ engine program will provide data on performance,

Table 2.6-1. Major Developmental Improvement Areas

- Concentrator
    - Weight reduction
    - Drag area
    - Packaging (launch)
    - Deployment versus erection
    - Surface slope error
    - Surface degradation
    - Micrometeoroid and debris

- Concentrator/Receiver Interface
    - Errors (alignment and pointing)
    - Concentration ratio (effect on concentrator facet size)
    - Aperture shield durability
    - Beam walk-off and walk-on

- TES Design
    - Configuration
    - TES heat transfer
    - Thermal conductance enhancement
    - Containment materials
    - Containment weight
    - TES utilization
    - Void management
    - Zero-g operation
    - One-g testing

- Receiver
    - Reduction of peak flux
    - Reduction of peak temperatures
    - Reduction of differential temperatures
    - Thermal cycling
    - Materials (creep/fatigue)
    - Heat pipe performance
    - Failure modes

- Stirling Engine Data Base
    - Performance
    - Design for solar temperature ratios (2.5-3.0)
    - Long term test experience
    - Efficiency versus weight
    - Code development

- System
    - Excess energy management

weight, and life; however, the program must also include the design and operational concerns particular to solar Stirling application. This includes higher temperature ratio operation and the need to provide for excess energy management.

Heat rejection and electric power conversion are not considered critical, although further development will be required in these areas also. Management of the excess energy resulting from seasonal and orbital variations is critical in the sense that the various design approaches can have a significant impact upon overall system design. The designs presented herein assume that the excess power is to be processed by the power system, resulting in excess electric power generation and excess waste heat to be rejected, thus requiring oversizing of various components such as the alternator, PLR, and radiator.

A recommended advanced technology program has been prepared, as the end product of this study, to promote development in these critical areas for solar dynamic power systems. Although a number of individual technology tasks have been identified, the tasks are naturally grouped by major subsystem. In most cases, several tasks would logically be combined into a single A/D program for the subsystem. The technology program development was carried through the following levels: identification of the technical issue and present SOA, a brief statement of work, benefits, impact if the technology is not developed, and technical risks involved.

## 2.7 SOLAR DYNAMIC POWER SYSTEM HARDENING

A determination of power system hardening was performed by an assessment of the survivability and an evaluation of the hardening potential of the solar dynamic power system designs developed during the study. A summary of the results is presented in section 13. The hardening study details are classified and are contained in reference 6. Nuclear weapons, lasers, neutral particle beam (NPB), and impact threats were considered in the study.

It was found that an order of magnitude improvement in laser and nuclear threat resistance could be designed into solar dynamic systems and that substantial impact resistance could be developed; however, there was a mass penalty of approximately 70% associated with such designs. It was also found that only a moderate degree of resistance could be designed into such systems as a defense against NPB weapons. Substantial impact resistance could be developed by the use of sacrificial design concepts and the use of redundancy for critical items. The inclusion of impact resistance accounts for a large percentage of the system mass increase. The development of detailed survivable solar dynamic power system designs will be dependent on the specific mission and orbit involved.

# 3.0 INTRODUCTION

## 3.1 BACKGROUND

The planned NASA Space Station has ushered in a new era of space power systems, with the advent of large photovoltaic solar arrays and large solar dynamic power systems for power generation. Future NASA, civil, commercial, and military missions will require space power systems of increased versatility and power levels. The NASA Lewis Research Center (LeRc) has initiated an Advanced Solar Dynamic (ASD) research project under the direction of NASA's Office of Aeronautics and Space Technology (OAST). The project is being implemented through a combination of NASA in-house efforts and contracted efforts. This study, the Solar Dynamic Power System Definition (SDPSD) Study is one of the contracted efforts. The SDPSD study addresses the key elements of the project: mission analysis to determine the power system requirements, system analysis to identify the most attractive ASD power systems to meet those requirements and to guide the technology development efforts, and technology development of key components.

The ASD project goal is to advance development of the ASD systems so as to realize the potential for efficient, lightweight, survivable, relatively compact, long-lived space power systems applicable to a wide range of power levels (3 to 300 kWe) and a wide variety of orbits. Successful development of these systems could satisfy the power needs for a wide variety of future missions, NASA and otherwise.

## 3.2 OBJECTIVES AND SCOPE

The SDPSD study objectives were identification of critical development areas for solar dynamic power systems, and recommendation of an advanced technology program to address these areas. The original study was divided into five technical tasks:

I. Definition of power requirements for future NASA, civil, and military missions

II. Study application and benefit of solar dynamic cycles to future mission requirements (beyond 1992 timeframe)

III. Conceptual design development and ranking of dynamic systems

IV. Identification of critical development areas and advanced concepts

V. Advanced technology program recommendations

Additional technical tasks were added to the contract:

VII. Evaluate the survivability level of the solar dynamic power system designs

VIII. Determined the hardening potential of the solar dynamic power system components

X. Update the Task II system characterization based on Task III results and current Phase B Space Station design data

XI. Develop a conceptual design of the integrated Stirling engine heat pipe heater head and the heat pipe solar receiver

Tasks VI and IX included the contract reporting requirements.

The final report format generally follows the order of Tasks I through V, with the following noted differences.

1. Task II was divided into two activities:
   - Selected missions
   - Characterization of solar dynamic power cycles, as updated by the Task X results
2. Task III was divided into two activities:
   - Conceptual designs of the solar dynamic power systems, including the Task XI integrated heat pipe Stirling engine
   - Ranking of the solar dynamic power systems

The scope of the SDPSD study was limited to evaluation of near-term SOA solar dynamic systems with Brayton, Rankine, and Stirling engine cycles. In addition, the dynamic systems were compared with photovoltaic systems; a SOA planar silicon cell (14.5% efficiency) array, and an advanced GaAs cell (22%

efficiency) concentrator array. The solar dynamic cycles were specified to be evaluated through the range of 1100K to 1400K (1520F to 2060F) peak cycle temperatures. For the Rankine cycle, this range of temperatures requires use of the alkali metals for the working fluid (cesium, potassium, etc.). The purpose of basing the designs on near-term SOA was to establish a technology base, to use in identification of critical development areas and advanced concepts.

An extensive data base was available to draw upon to support this study as a result of other previously performed or presently ongoing power related programs. The first of these was the design work for the CBC and ORC cycles prepared for the Space Station Phase B study effort (ref. 1). Design data for the free piston Stirling engine was obtained from the NASA-LeRC Stirling Engine Office. Additional Stirling engine data for a range of power levels was provided by Sunpower, Inc., under a subcontract to this study (ref. 5). Results of three A/D programs recently performed in support of the Space Station program were also utilized: concentrator work by Harris Corporation (ref. 3), heat pipe radiator work by Grumman Aerospace Corporation (ref. 2), and solar receiver work by Boeing Aerospace Company (L.M. Sedgwick, Boeing, Contract NAS3-24669). Liquid metal Rankine cycle design data was obtained from the Sundstrand Corporation ASTEC program (ref. 4), performed in the 1960s. This was a design and experimental program for a 15 kWe solar dynamic Rankine cycle using rubidium as the working fluid.

As a prime contractor for Space Station WP-04, Rocketdyne established a team relationship with major suppliers of both solar dynamic and photovoltaic power subsystems. These suppliers, including Garrett, Sundstrand, and Ford Aerospace, provided additional data to assist in the conduct of this study program.

## 3.3 PROGRAM APPROACH

The mission selection task relied on the mission data base maintained by the Rockwell International Space Transportation Systems Division. Missions and power levels were then selected and submitted to NASA for approval. Conduct of the analytical study characterizing the Brayton, Rankine, and Stirling cycles utilized a data base representing Space Station technology levels and solar dynamic power system codes (which had been developed on company funds) updated to meet the needs of this program. Each power system was optimized for peak cycle temperatures, as well as optimization of numerous cycle parameters such as concentration ratio, radiator size, engine parameters, etc.

Two types of photovoltaic arrays were also sized for comparison to the solar dynamic cycles, the silicon array being based on Space Station technology levels. Preliminary optimization for the cycles was on the basis of power system weight and area. Conceptual designs were then prepared for each cycle at each of the power levels.

A formal ranking of the solar dynamic power systems was then performed encompassing a number of power system characteristics besides weight and area. Finally, critical development areas were identified and a recommended advanced technology program prepared.

## 3.4 STIRLING ENGINE DESIGN DATA SUBCONTRACT

In the course of the study, it was found necessary to obtain more information regarding Stirling engine design, size, and performance. A subcontract was let to Sunpower, Inc. to extend their 25 kWe, temperature ratio = 2.0, single-cylinder free piston space power module design to the power levels of 8 kWe and 40 kWe gross power output from the alternator. The design data was prepared by Gedeon Associates, in consultation with Sunpower (ref. 5).

The space power module (ref. 5) engine heater was designed for a liquid metal pumped loop heat supply. Subsequently, Sunpower and Mechanical Technology Incorporated (MTI) collaborated on the design of a heat pipe heater head for the 25 kWe Stirling Space Engine (SSE). Task XI of this study required design of an integrated heat pipe heater Stirling engine with a heat pipe solar receiver and including integral thermal storage.

## 4.0 DEFINITION OF POWER REQUIREMENTS FOR FUTURE NASA MISSIONS

The work reported in this section corresponds to Task I from the Statement of Work (SOW).

### 4.1 OBJECTIVE

Future NASA missions of beyond 1992 shall be reviewed and solar dynamic power system application requirements for these missions shall be identified. A power system range of 3 to 300(+) kWe shall be considered. Since the power conversion requirements can be generic, missions with nuclear heat sources shall be considered with the focus on dynamic power conversion system requirements. Earth orbital, interplanetary (near sun) orbit transfer, and potential lunar missions shall be considered. Orbital missions shall include equatorial and polar orbits and altitudes ranging from LEO, through intermediate, to GEO. Growth space station requirements shall be considered. Power system requirements for commercial spacecraft shall be included. A listing of the missions and the mission power system requirements shall be identified and prepared.

### 4.2 SCOPE OF THE MISSION ANALYSIS

A review of future NASA, military, and commercial missions clearly shows a trend to higher power requirements. To the present, space power needs have been met with photovoltaic systems with relatively low power, 10 kWe or less. Solar dynamic power systems offer the potential for higher power, lightweight, highly reliable, long-lived systems that can survive in a variety of altitudes and orbits. For LEO missions, the substantially reduced drag area for solar dynamic versus photovoltaic power systems is a major benefit in terms of orbit maintenance requirements.

The effort devoted to this task of definition of mission power requirements was purposefully small, as the task was intended as a means to the end of selecting two representative missions and power levels upon which the balance of the study would be based. As such, it was not necessary that the mission analysis be exhaustive, but rather sufficiently representative of future mission requirements. Rockwell's in-house mission data base was chosen as being quite adequate for this task.

## 4.3 ROCKWELL MISSION DATA BASE

Rockwell's Space Transportation Systems Division (STSD), builder of the National Space Transportation System (NSTS) - the Space Shuttle, has created and maintains an extensive space mission information data base. STSD was a contractor for the Space Transportation Architecture Study (STAS), which utilized a mission model data base provided jointly by NASA and DOD (Civil Missions Data Base, Revision #6, NASA, 18 October 1985). STSD has a variety of data sources in addition to the STAS mission model, both computer based and in hard copy form, as listed in table 4.3-1.

A more recent study (D.H. Herman, Chairman, Civil Applications of Space Nuclear Power - Final Report of the Civil Missions Advisory Group, NASA Headquarters, August 1984) identified a number of missions in the 100 to >1000 kWe power range, as shown in table 4.3-2. These missions could be accomplished with a nuclear power source; however, ASD systems do provide a non-nuclear alternative.

The STAS mission model includes Space Station, Space Station co-orbiters and free flyers, commercial systems, lunar base, and GEO station. Other data bases were used to review military missions (classified), interplanetary, large observatories, and GEO COMSATS. An example of a portion of the mission power data obtained from the STAS mission model is shown in table 4.3-3. The full range of missions considered for this study was shown previously in figure 2.2-1.

The constraints used to search the mission data base were power level of 3 to 300 kWe, scheduled in the time frame of 1992 to 2010, and excluding missions associated with the initial Space Station. The missions obtained from the STAS mission model were grouped in two ways: by power level as shown in figure 4.3-1, and by mission type (orbital location) as shown in figure 4.3-2. Many of the missions examined are destined for the (growth) Space Station and would draw power from the station, and as such are not directly useful in determination of stand-alone power system sizing. This data base only identified four missions in the 25 to 300 kWe power range. Several more of the higher power missions were found in the other data bases.

Table 4.3-1. Sources of Space Mission Information

- NASA PLANNING DOCUMENTS
  - POPs
  - NASA 5-YEAR PLANS
  - SSEC PLANETARY PROGRAM STUDY
  - SPACE STATION (LANGLEY) MISSION MODEL
  - STAS MISSION MODEL
  - MSFC MISSION MODEL
  - NASA TECHNOLOGY MODELS
- DOD SPACE MISSION MODELS
  - STAS MISSION MODEL
  - BUDGET SUBMISSIONS
- BATTELLE "OUTSIDE USERS" PAYLOAD MODEL
  - NON-NASA, NON-DOD PAYLOADS
  - INCLUDES ARIANE AND FAR EASTERN (CHINA/JAPAN) PAYLOADS
- ROCKWELL MISSION ANALYSES AND MARKET SURVEYS
  - NASA, DOD, FOREIGN AND COMMERCIAL MARKETS AND MISSIONS
  - CONTRACT STUDIES (SDI, SPACE STATION, ETC.)
- OTHER GOVERNMENT PLANNING DOCUMENTS
  - BUDGETARY REQUESTS, CONGRESSIONAL TESTIMONY, ETC.
- CURRENT LITERATURE AND PERSONAL CONTACTS

Table 4.3-2. Missions Identified by Civil Missions Advisory Group

| Mission | Orbit | Mission Date, Year | Power Level, kWe |
|---|---|---|---|
| Manned orbital facility | | | |
| Initial space station | LEO | 1990 to 2000 | 75 to 150 |
| Growth space station | LEO | 2000 to 2010 | 300 to 500 |
| Advanced space station | LEO | 2000 to 2010 | 500 to 1000 |
| Earth science and applications | | | |
| GEO communications platform | GEO | 1990 to 2000 | 100 to 200 |
| Air/ocean traffic control | LEO | 2000 to 2010 | 100 to 200 |
| Transportation | | | |
| GEO payload delivery | LEO-GEO | 1990 to 2000 | 100 to 200 |
| Lunar payload delivery | LEO-lunar | 2000 to 2010 | 100 to 200 |
| Manned Mars mission | IP* space | Beyond 2010 | > 1000 |
| Asteroid base resources | | | |
| Material processing | IP space | Beyond 2010 | 200 to 500 |
| Planetary exploration | | | |
| Multi-asteroid sample return | IP space | 2000 to 2010 | 100 to 200 |
| Comet nucleus sample return | IP space | 1990 to 2000 | 100 to 200 |

*NOTE: IP-Interplanetary

## Table 4.3-3. Sample Mission Model Data

| DIS. | CONNECTED PROGRAMS | HQ NUM | HQ CODE | | MISSI TYPE | DEST | PROGRAM NAME | <<<<POWER(KW)>>>> TYPE | OPER | HOURS | PEAK | HOURS | ORBIT INCL (DEG) | ORBIT ALT (NM) | LAUNCH SIZE P/L (LB) | L (FT) | W/D (FT) | H (FT) | RET P/L (LB) |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| A | | 1000 | E | US | | 1&3 | ADVANCED SOLAR OBSERVATORY | DC | 5 | 16 | 7.5 | 1 | 28.5 | 270 | 27560 | 26.9 | 12.5 | NA | NA |
| A | | 1000.5 | E | | CC | | ADVANCED SOLAR OBSERVATORY | | | | | | 28.5 | 270 | 3307 | ND | ND | ND | NA |
| A | | 3000 | E | J | | 6 | ASTRONOMICAL PLATFORM | DC | 5 | 21 | 7 | 3 | 28.5 | 220 | 4410 | 16.4 | 13.1 | NA | NA |
| A | | 3002 | E | | CC | | ASTRONOMICAL PLATFORM | | | | | | 28.5 | 220 | 6615 | ND | ND | ND | 6615 |
| A | | 3001 | E | | A | | ASTRONOMICAL PLATFORM | | | | | | 28.5 | 220 | NA | NA | NA | NA | 4410 |
| A | | 3003 | E | | S | | ASTRONOMICAL PLATFORM | | | | | | 28.5 | 220 | 395 | ND | ND | ND | NA |
| A | | 3015 | E | J | | 3 | LINE GAMMA DETECTION | DC | 2 | 22 | 3 | 2 | 28.5 | 270 | 1100 | 3.3 | 3.3 | 3.3 | NA |
| A | | 3016 | E | | A | | LINE GAMMA DETECTION | | | | | | 28.5 | 270 | NA | NA | NA | NA | 1100 |
| A | | 3017 | E | | S | | LINE GAMMA DETECTION | | | | | | 28.5 | 270 | 265 | ND | ND | ND | 0 |
| EO | | 3060 | E | ESA | | 8 | MORNING PLATFORM | DC | 5 | 20 | | | 98 | 420 | 4190 | ND | ND | ND | NA |
| EO | | 3061 | E | | CC | | MORNING PLATFORM | | | | | | 98 | 420 | 660 | ND | ND | ND | NA |
| EO | | 3035 | E | ESA | | 8 | AFTERNOON PLATFORM | DC | 4.7 | 20 | | | 98 | 420 | 3750 | ND | ND | ND | NA |
| EO | | 3036 | E | | CC | | AFTERNOON PLATFORM | | | | | | 98 | 420 | 660 | ND | ND | ND | NA |
| EO | | 3063 | E | J | | 3 | OBSERVATIONS OF UPPER ATMOSPHEAC | | 5 | 2 | | | 28.5 | 270 | 505 | 6.6 | 11.5 | 9.8 | NA |
| EO | | 3064 | E | | A | | OBSERVATIONS OF UPPER ATMOSPHERE | | | | | | 28.5 | 270 | NA | NA | NA | NA | 505 |
| EO | | 3068 | E | J | | 8 | RTRSOC | AC | 4 | 23.8 | 5 | 0.2 | 98 | ERR | 4410 | 39.4 | 13.1 | NA | NA |
| EO | | 3039 | E | J | | 7 | EARTH OBSERVING FACILITY | DC | 3 | 24 | | | 98 | ERR | 5510 | 13.1 | 13.1 | NA | NA |
| EO | | 1115 | E | US | | 4 | LIDAR FACILITY | DC | 4.5 | 24 | | | 28.5 | 270 | 3085 | 8.2 | 8.2 | 8.2 | NA |
| EO | | 1116.5 | E | | CC | | LIDAR FACILITY | | | | | | 28.5 | 270 | 882 | ND | ND | ND | 882 |
| EO | | 1116 | E | | A | | LIDAR FACILITY | | | | | | 28.5 | 270 | NA | NA | NA | NA | 4190 |
| EO | EOS | 1122.5 | E | US | | 8 | MOD RES IMAGING SPECTROMETER-TDC | | 0.2 | 10 | 0.275 | 0.03 | 98 | 380 | 220 | 2 | 1 | 0.8 | NA |
| EO | EOS | 1122 | E | | CC | | MOD RES IMAGING SPECTROMETER-T | | | | | | 98 | 380 | 55 | ND | ND | ND | 55 |
| EO | EOS | 1102.5 | E | | | 8 | HIGH RESOLN IMAGING SPECTRMTM | DC | 0.825 | 2 | | | 98 | 380 | 2646 | 9.2 | 6.6 | 6.2 | NA |
| EO | EOS | 1102 | E | | CC | | HIGH RESOLN IMAGING SPECTRMTM (HIRIS) | | | | | | 98 | 380 | 165 | ND | ND | ND | 165 |
| EO | EOS | 1101.5 | E | | | 8 | HIGH RES MULTIFREQ RADIOMETER | DC | 0.32 | 24 | | | 98 | 380 | 705 | 19.8 | 13.2 | 25.4 | NA |
| EO | EOS | 1101 | E | | CC | | HIGH RES MULTIFREQ RADIOMETER (HRMR) | | | | | | 98 | 380 | 595 | ND | ND | ND | 595 |
| EO | EOS | 1114.5 | E | | | 8 | LASER ATMOS SOUNDER & ALT-A | DC | 2.5 | 12 | 3 | | 98 | 380 | 4520 | 14 | 6.6 | 6.6 | NA |
| EO | EOS | 1114 | E | | S | | LASER ATMOS SOUNDER & ALT-A | | | | | | 98 | 380 | 440 | ND | ND | ND | 440 |
| EO | EOS | 1150.5 | E | | | 8 | SYNTHETIC APERATURE RADAR | DC | 4 | 4.8 | 9.5 | 1.2 | 98 | 380 | 3087 | ND | ND | ND | NA |
| EO | EOS | 1150 | E | | CC | | SYNTHETIC APERATURE RADAR | | | | | | 98 | 380 | 660 | ND | ND | ND | 660 |
| EO | EOS | 1077.5 | E | | | 8 | ALTIMETER | DC | 0.2 | 24 | | | 98 | 380 | 331 | 5 | 5 | 3.3 | NA |

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Table 4.3-3. Sample Mission Model Data (Concluded)

Definition of Terms

| | |
|---|---|
| DIS. | discipline (type of mission) |
| A | astronomy and astrophysics |
| EO | earth observation (general grouping) |
| EOS | earth observation systems (a specific EO group) |
| NUM | mission identification number |
| HQ CODE | NASA Headquarters code |
| E | code E |
| (Blank) | originating country/group |
| US | USA |
| J | Japan |
| ESA | European Space Agency |
| MISSI TYPE | mission type |
| CC | configuration change during mission |
| A | payload to be returned to earth |
| S | servicing during mission |
| DEST | identification as to type/location of orbit |
| POWER (KW) | mission power requirements |
| TYPE | ac or dc power |
| OPER HOURS | power (kWe) and hours per day |
| PEAK HOURS | peak power (kWe) and hours per day |
| ORBIT | mission orbit |
| INCL | inclination, degrees |
| ALT | altitude, nmi |
| LAUNCH SIZE | mission payload |
| P/L | weight, lb |
| L | length, ft |
| W/D | width or diameter, ft |
| H | height, ft |
| RET P/L | weight to be returned to earth, lb |
| NA | not applicable |
| ND | not defined |

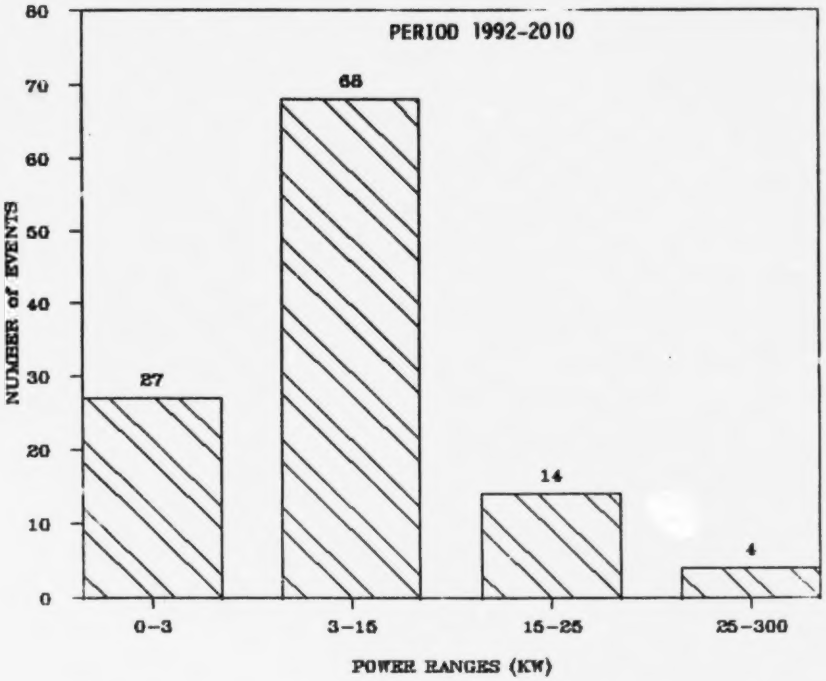


Figure 4.3-1. Distribution of Power Requirements

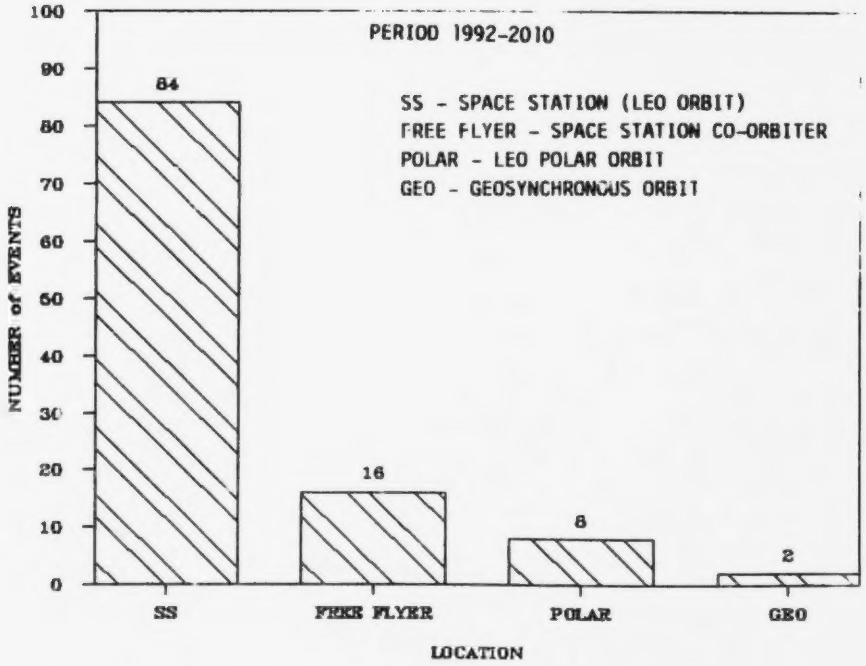


Figure 4.3-2. Orbital Location of Power Requirements

## 4.4 MISSION ANALYSIS

The following groups of missions were examined for the mission analysis task, representing a total of over 100 missions:

- Growth Space Station
- Space Station Co-Orbiters
- Commercial Systems
- Earth Observation Systems
- Earth Sensing Systems
- Lunar Base
- DOD Missions
- SDI Missions
- Interplanetary
- Large Observatories
- Growth COMSATS
- GEO Station
- GEO Platforms

Design requirements definition for the various spacecraft missions is in a state of evolution; therefore, it may be expected that the requirements will change somewhat in the future. The SOW specified preparation of a listing of the missions including: type of mission and date, orbital characteristics and altitude, power level and electrical output characteristics, lifetime and reliability, launch and deployment, and interdependence of the spacecraft and its power system. In many cases, much of the mission and spacecraft information is yet to be defined.

Partial summaries of mission analysis results may be found on figures 4.3-1 and 4.3-2, and in table 4.4-1, which show that at present, many more missions fall below 25 kWe than above. Presentation of the detailed mission analysis results is organized in the following subsections by missions groups, as listed above. The results have been summarized to include only mission type, date, orbit, and power.

### 4.4.1 Growth Space Station

The initial Space Station missions were excluded from this study. The growth version of the Space Station will incrementally increase power in subsequent years, from the original 75 kWe to 300-400 kWe. Power growth will be incremental by incorporation of pairs of 25 kWe solar dynamic modules, possibly of progressively more advanced designs, to the initial photovoltaic

Table 4.4-1. Missions Developed Under Systems Definition Study

| Mission | Orbit | Mission Date, Year | Power Level, kWe |
|---|---|---|---|
| Materials Processing | | | |
| Micro-gravity variable | LEO | 1998 | 50 |
| Automated materials processing | LEO | 1996 | 10 |
| Materials processing lab/Canadian | LEO | 1996 | 20 |
| Commercial space processing | LEO | 1996 | 5.5 |
| Life Science | | | |
| Production bio-processing | LEO | ? | 7 |
| Biological production units | LEO | 1996 | 16 |
| General purpose research/European | LEO | 1992 | 10 |
| Medical experiments technology | LEO | 1996 | 5 |
| Earth Observation Systems | | | |
| LASA-B | Sun sync. | 2000 | 7.8 |
| Doppler LIDAR | Sun sync. | 1996 | 3 |
| Synthetic aperture radar | Sun sync. | 1996 | 4 |
| Earth resources sensing | LEO | 1999 | 10 |
| Observation of upper atmosphere/Japanese | LEO | ? | 5 |
| Ice-Earth monitoring radar/Canada | LEO-polar | ? | 4 |
| Earth/Sun Interaction | | | |
| Solar terrestrial observatory | GEO-polar | 1993 | 6 to 10 |
| Communication | | | |
| Large platforms | GEO | 1996 | 10 to 30 |

solar power array. The missions aboard the Space Station will take their power from a central power distribution system, so the individual missions do not designate a need for specific solar dynamic power system sizes. Nonetheless, since these missions could be shifted to co-orbiters, the missions were individually included in the mission analysis survey.

The Space Station nominal orbit will range between 180 and 270 nmi (333 and 500 km) altitude at 28.5° inclination. The growth station will see transition from experimental to commercial materials processing, larger telescopes, use as a transportation node for other space assets, and will see expanded human involvement. The missions considered were:

**Materials Processing**

| | | |
|---|---|---|
| 1996-2010 | Micro-g and materials processing | 60 kWe |
| 1999 | Materials science research lab | 15 kWe |

**Life Sciences**

| | | |
|---|---|---|
| 1999 | Closed environmental life support system | 30 kWe |
| 1996 | Biological production units | 16 kWe |

**Other**

| | | |
|---|---|---|
| 1998 | Tethered fluid storage | 10 kWe |

### 4.4.2 Space Station Co-Orbiters

The co-orbiting platforms are free-flyers maintained in the vicinity of Space Station. They meet the need for long-term steady g-level (zero-g), ready access to the station for servicing, and periodic access to the Space Shuttle for transport of processed materials. The missions considered were:

**Materials Processing**

| | | |
|---|---|---|
| 1998 | Micro-g, variable | 50 kWe |
| 1996 | Automated materials processing | 10 kWe |
| 1996 | Materials processing lab | 20 kWe |
| 1996 | Commercial space processing | 5.5 kWe |

**Life Science**

| | | |
|---|---|---|
| 2000+ | Production bio-processing | 7 kWe |

**Other**

| | | |
|---|---|---|
| 2000 | Low acceleration propulsion | 5 kWe |
| 2000+ | Astronomical platform | 5 kWe |
| 1995 | Contained plasma experiment | 5 kWe |
| 1996 | LIDAR facility | 4.5 kWe |

### 4.4.3 Commercial Systems

This mission grouping includes commercial processing of materials in space for use on Earth. The grouping was made to separate commercial missions from science and technology development missions since the assignment of commercial missions between Space Station and co-orbiters has yet to be decided. The missions considered were:

Materials Processing

| | | |
|---|---|---|
| 1996 | Crystal production units | 32 kWe |
| 1996 | Material processing development | 20 kWe |
| 1999 | ECG (semiconductor crystal) production unit | 20 kWe |
| 1992 | Electrophoresis operations | 15 KWe |
| 1996 | Containerless process production | 8 kWe |

Life Science

| | | |
|---|---|---|
| 1996 | Biological production units | 16 kWe |

## 4.4 Earth Observation and Earth-Sensing Systems

These missions provide continuous observation of terrestrial and nearby space environment. The missions include land, sea, atmospheric, and geomagnetic observations, agriculture, forestry, and fisheries monitoring, resource identification, plasma/atmospheric interactions, etc. The Earth observation mission orbits are usually sun-synchronous, which is near-polar and at LEO altitudes. The Earth observation missions considered were:

Sun-Synchronous Platforms

| | | |
|---|---|---|
| 2000 | LASA-B | 7.8 kWe |
| 1996 | Synthetic aperture radar | 4 kWe |
| 1996 | Doppler LIDAR | 3 kWe |

Other

| | | |
|---|---|---|
| 1995 | Medium resolution imaging radiometer | 26 kWe |
| 1994 | Multispectral linear arrays | 24 kWe |
| 1994 | Synthetic aperture radar | 24 kWe |

The high inclination LEO orbits have the potential to be technology drivers for the area of excess energy management. That is, orbits which have no eclipse and maximum solar intensity may have 1.5-1.7 times the solar exposure in comparison to the design orbit with maximum eclipse and minimum solar intensity; for Space Station the ratio is 1.22.

The Earth-sensing mission orbits are located at a variety of orbit inclinations and mostly at LEO altitudes. The Earth-sensing missions were:

| | | |
|---|---|---|
| **Terrestrial** | | |
| 1999 | Earth resources sensing | 10 kWe |
| 1996 | $CO_2$ LIDAR, wind and trace gases | 10 kWe |
| 2000+ | Observations of upper atmosphere | 5 kWe |
| 1992 | Morning/evening platform | 5 kWe |
| 2000+ | Radars/sensors, station and polar | 4 kWe |
| 1998 | Earth observation facility | 3 kWe |
| **Earth/solar interaction** | | |
| 1993 | Solar terrestrial observatory | 6-10 kWe |

### 4.4.5 Lunar Base

The lunar base is a manned base on the lunar surface for scientific exploration, resource utilization, and colonization. Such a base is a potential source for metals, semiconductor material, propellant, and shielding material. The base would be an excellent location for astronomical telescopes. The lunar base could be the next big NASA project after Space Station, and would no doubt involve international cooperation.

An example of lunar mining would be hydrogen reduction of lunar soil to produce oxygen (sized for 3400 lb/24 hours) to provide propellant for the transportation system supporting the base. The oxygen production would begin with magnetic separation of ilmenite ($FeTiO_3$) from the lunar soil, and production of titanium oxide, iron, and oxygen. The process requires both thermal and electrical energy.

Base power requirements could range up to 400-500 kWe and 150+ kWt. The lunar base mission is one-of-a-kind and involves 14 Earth-day sunlight and darkness periods, and hence does not appear to be a feasible application for a solar dynamic system. It may not be practical to store enough thermal energy to provide power for the long period of darkness.

### 4.4.6 DOD and SDI Missions

These missions are generally classified, hence few details are available as to power requirements. The DOD missions categories are navigation, observation, early warning, weather, and communications. Power requirements vary widely for these types of missions, with the highest requirement in the 5-10 kWe range.

The SDI missions are designed for strategic defense against ICBMs, and the architecture requirements are as yet being established. Most SDI missions will involve three levels of power associated with the platform operational status: station keeping, alert, and battle mode. Power may range from as little as 1 kWe up to 100s of MWe.

Applicability of solar dynamic power to the DOD and SDI missions is presently uncertain, although solar dynamics may offer survivability advantages over photovoltaic arrays, smaller areas, and higher power levels.

### 4.4.7 Interplanetary

A total of nine interplanetary science probe missions were identified:

- Galileo
- VRM (Venus Radar Mapper)
- Mars Observer
- Lunar Observer
- Near-Earth Asteroid Rendezvous
- Mars Aeronomy Observer
- Comet Rendezvous/Asteroid Flyby
- Cassini (Saturn Orbiter/Titan Probe)
- Main Belt Asteroid Rendezvous

All of these missions are based on either the Mars Orbiter power system which uses solar arrays, or on the Mariner Mark II power system which is baselined for radioisotope thermoelectric generators (RTGs). None of the missions exceed 1 kWe power; therefore, the interplanetary planetary missions were excluded as being too low in power for this study.

### 4.4.8 Large Observatories

There are several free-flying astronomical observatories in various stages of evolution (some in a pre-concept phase). The missions considered were:

| | | |
|---|---|---|
| 1986 | Space Telescope (photovoltaic), built | 2.1 kWe |
| 1988 | Gamma Ray Observatory (photovoltaic), in construction | 1.5 kWe |
| 1992/3 | Advanced X-Ray Astrophysical Facility, definition phase | 2.0 kWe |
| 1993/4 | Space Infrared Telescope Facility, definition phase | 2.0 kWe |
| 2000+ | Large Deployable Concentrator, concept phase | 3.5 kWe |
| 2000+ | Large Area Modular X-Ray Telescope, pre-concept phase | 1.1 kWe |
| 2000+ | 100 m Thinned Aperture Telescope, pre-concept phase | 25 kWe |
| 2000+ | COSMIC (Coherent Optical System of Modular Imaging Collectors), pre-concept phase | 25+ kWe |

The observatories will normally be in the plane of the Space Station to allow periodic servicing, although at an altitude of 300-400 nmi (556-741 km), thus avoiding much of the effects of drag and atomic oxygen. For this study, only the latter two missions were considered as being within the required range of power (3 to 300 kWe).

### 4.4.9 Growth COMSATS

The field of commercial communication satellites (COMSATS) will continue to grow, both in number and in size, as the need for communication channels expands world-wide. Systems are currently entering the 3+ kWe power range.

Competition plus the need to provide economical user rates makes production and operation of the satellites sensitive to both price and technology. Long life and high reliability are important drivers.

COMSAT power has been, and will be for some years more, provided by photovoltaic power. A number of the COMSATS built in the past were examined as to power requirements, and the highest power cases were found to be 3 kWe for the Fordsat (1984) and 5 kWe for the Galaxy DBS (1986). Projecting the power growth to the 1992-2010 time frame, it is probable that design power levels may enter the range of perhaps 10-15 kWe, well within the range of power to be considered for this solar dynamic power study.

### 4.4.10 GEO Station

The GEO Station is a Space Station-derived manable facility in GEO. It could serve as a base for OTVs (orbit transfer vehicles) and OMVs (orbital maneuvering vehicles), a spare parts warehouse, and/or a fuel depot. It would serve as a habitat during periodic visits of manned OTVs originating from LEO Space Stations. This project is in the pre-concept phase and is estimated to need 25-35 kWe power.

Power requirements for this project were interesting because in GEO orbit, the station is exposed to continuous sunlight for months. The eclipse periods, when they do occur, are about twice as long as for LEO. In addition, required power for periods of housekeeping and during periods of habitation will be different. Information subsequent to the mission analysis indicates that the GEO Station would be beyond the 1992-2010 time frame of this study.

### 4.4.11 GEO Platforms

The GEO Platform program is in the concept phase, with the potential for multiple similar platforms. The platforms will be large, weighing over 10,000 lb, and would fulfill both multipurpose and dedicated roles with both U.S. government and industry involved in development and utilization. Physical size of the platform and power system would require deployment or assembly in space, possibly in LEO with subsequent placement in GEO. Power requirements for the platform are estimated as 10-30 kWe, and potentially higher.

## 5.0 SELECTION OF REPRESENTATIVE MISSIONS

The work reported in this section corresponds to the mission selection activity of Task II from the SOW.

### 5.1 OBJECTIVE

The missions identified in Task I shall be classified into two technology classes; 3 to 25 kWe and 25 to 300 kWe. Missions shall be selected that represent a broad range of lifetimes and orbital characteristics. A representative mission and power level then shall be selected from each category for study and evaluation with the solar Brayton, solar Rankine, and solar Stirling engine cycles. The selected missions and power levels shall be submitted to the NASA Project Manager for approval. Upon approval, an analytical study shall be conducted, evaluating the application of each dynamic cycle to each selected mission.

### 5.2 MISSION SELECTION CRITERIA

The mission selection process required an assessment of which of the mission parameters would have an influence on design, operation, and reliability of a solar dynamic power system. There are a number of mission variables which may be considered in mission selection:

- Orbital characteristics (polar, equatorial, elliptical)
- Orbit altitude (LEO, intermediate, GEO)
- Power level
- Generic nature of mission and power requirements
- Reliability and design operational life
- Serviceable versus unserviceable missions
- Natural environmental concerns
    - Van Allen belt radiation (peaks at 3000 and 16,000 km)
    - Atomic oxygen (degradation of materials)
    - Micrometeoroids and debris (varies with altitude)

The stated end purpose of this study was to recommend an advanced technology plan for critical development areas of solar dynamic power systems. To base mission selection upon the choice of one or two specific planned future missions does place a reliance on the perception of the mission planners as to the characteristics of future power systems (power, weight, area, reliability, cost, etc.). Of course, a secondary purpose of this study was to assist in clarifying perceptions in regards to solar dynamic power systems characteristics. Early in the mission analysis, it became obvious that choice of missions based on population would not necessarily push technology issues to the fore, as should be the case to achieve the end purpose of the study.

An earlier chart, figure 4.3-1, indicated the distribution of power requirements based on the STAS mission model, excluding initial Space Station missions. Few missions required more than 25 kWe and only two were located beyond LEO altitudes. Summarizing the missions discussed in section 4.4 (a combination of typical STAS missions and other missions) leads to much the same conclusions. The wide range of power requirements might be grouped as follows:

| | Power, kWe | Number |
|---|---|---|
| LEO Missions | 3-10 | 20 |
| | 15-26 | 12 |
| | 30-32 | 2 |
| | 50-60 | 2 |
| GEO Missions | 10-15 | * |
| | 10-30 | 1 |
| | 25-35 | 1 |

*Note: There will be many growth COMSATS with various power requirements.

Based on the number of missions, the tendency would be to select missions in LEO orbital altitude. LEO altitude would also be selected on the basis of power system thermal cycling due to eclipse, since LEO orbits experience about 60 times as many eclipse cycles per year as do GEO orbits. Thermal cycling imposes stresses upon system components which must be accounted for in power system design. Thus, LEO missions emphasize the issue of fatigue much more than do the GEO missions.

Considering that the power conversion unit (PCU) will run continuously at constant power conditions, then the only operational difference between LEO and GEO orbits will be thermal management by the concentrator/receiver/storage subsystems. In a 500 km altitude orbit, about 1.6 times as much energy must be absorbed by the receiver during sunlight as is immediately used by the PCU, the remainder going to thermal storage for use during the 36-minute (maximum) eclipse period. In GEO, this factor is about 1.05 for a 69-minute (maximum) eclipse time. Thus, the thermal design differences between LEO and GEO are:

- The concentrator is larger in LEO (1.6/1.05)
- The receiver is larger* in LEO (1.6/1.05)
- Thermal storage is larger in GEO (69/36)

*Note: For designs with thermal storage integral with the receiver, the receiver would be larger in GEO, as stored thermal energy must be nearly doubled for the GEO orbit.

The requirements for power system operation are similar except that in LEO thermal energy is stored for 62 minutes and discharged for 32 minutes on average (design values are 59/36 minutes). For the case of maximum eclipse in GEO, thermal energy is stored for 22 hours, 51 minutes and discharged for 69 minutes (design values). However, the GEO orbit experiences eclipse only during the equinoxes, for a total of 90 eclipses per year.

The natural environmental concerns are primarily orbital dependent. Degradation of materials due to atomic oxygen interaction at LEO altitude is a significant design concern. The primary hazard in and near the Van Allen belt is the effect of charged particles upon electronics, both power electronics and control electronics. The power electronics ordinarily are required to handle large current and voltages, so are less susceptible to upset, whereas control electronics operate ordinarily with milliamps and $\leq$10 volts. Because of this, the quantity of spurious energy required to produce a problem in control electronics is orders of magnitude smaller than that required to affect other spacecraft components, including power electronics.

The matter of whether a spacecraft (mission) may be serviced periodically or not influences design of the power system. Service may be performed as planned maintenance and resupply, or unplanned repair. Such service could be accomplished by visiting the spacecraft in orbit, or by retrieval of the spacecraft, and servicing may be either robotic or man-assisted.

A brief comparison of photovoltaic power systems to solar dynamic power systems indicated many similar concerns, such as: thermal cycling, energy storage, component degradation, atomic oxygen protection, etc. Comparing GEO to LEO applications, the weight and area advantages indicated for solar dynamic over photovoltaic (see section 2.3.1) would be somewhat smaller for the GEO orbit. The major factors are energy storage (battery) weight, and solar array area and weight. The relatively few eclipse cycles experienced in GEO allows 70-80% battery depth of discharge versus ≤35% for LEO. Thus, in spite of the nearly double eclipse interval, battery weight would be the same or less for GEO, whereas energy storage weight must be nearly doubled for solar dynamic systems for GEO. However, the energy storage weight savings for photovoltaic would be partially offset by an increase in solar array area and weight required to compensate for higher radiation damage which would occur in GEO orbit.

The foregoing assessment established the mission criteria which will have the greatest effect upon solar dynamic power system design, operation, and reliability:

- Orbital location (altitude and inclination)
    - Thermal cycling/fatigue
    - Size of concentrator/receiver/storage
    - Natural environmental effects
- Power level
- Serviceability

## 5.3 SELECTED MISSIONS/POWER LEVELS

A different representative mission was chosen for each power category (3 to 25 kWe and 25 to 300 kWe). The missions are:

| Mission | Altitude | Inclination | Net Power | Excess Energy Ratio | Service |
|---|---|---|---|---|---|
| 1 | 500 km<br>(270 nmi) | 28.5° | 35 kWe | 1.22 | Serviceable |
| 2 | 1200 km<br>(648 nmi) | 60° | 7 kWe | 1.57 | Unserviceable |

A summary of the rationale for the selection follows.

Considering an engine with 50 kWe capability and an efficiency of 33%, the concentrator/receiver/storage device would have to supply 150 kWt of thermal power continuously to the engine. For LEO, the receiver would have to absorb over 240 kWt from the concentrator during sunlight, and the storage device would have to absorb the excess needed for eclipse (150 kWt * 0.6 hr = 90 kWhr thermal). In GEO, these devices would be capable of supplying energy to drive a 76 kWe engine (based on receiver power), although the storage device developed would have to be increased in capacity (modularly) to support the larger engine and longer eclipse duration for GEO. As a result of the foregoing concern as to concentrator/receiver sizing and the frequency of eclipse cycling, it was decided that one mission would be in LEO, and at the same inclination as Space Station, 28.5°.

Consultation with the NASA Project Manager resulted in a preference that one mission be a generic type of military mission, in the 5-10 kWe range, and be operated nearer the Van Allen belt, with orbit inclination variable from 0-90°. LEO orbits with inclinations of approximately 30° to 90° have the potential of no eclipse (depending on altitude) thereby requiring that the solar dynamic system design be capable of dumping up to 35-40% of the energy available at maximum solar input as a result of the absence of an eclipse interval. (A system operating in GEO must also be capable of dumping heat continuously, but at a much lower rate, approximately 5% of the energy available.)

For military application, the altitude range of 900-1500 km is being considered, which is below the severe Van Allen belt radiation altitude, found at 3000 km. The 5-10 kWe power level for the 900-1500 km mission satisfied the lower specified power category of 3 to 25 kWe. Therefore, the lower altitude 500 km mission was assigned to the 25 to 300 kWe power category.

The higher power level was selected as 35 kWe net power output. This power is somewhat above the 25 kWe split between power categories, and it is approximately 22% greater than the Phase II Space Station solar dynamic module, when compared on a gross power output basis (about 39 versus 32 kWe). On the basis of higher cycle performance due to increased operating temperature, much of the Space Shuttle launch sizing considerations for Space Station also applied to the 35 kWe power systems.

The lower power level was selected as 7 kWe net power output. Therefore, the power ratio for the selected missions was an even 5:1. Power conversion was required in all cases, whether from the solar dynamic or photovoltaic power systems, to produce a common net output power condition of frequency and voltage.

Although the missions do not represent specific missions taken from the data base examined in Task I, they do represent the power population encountered in Task I. The possible number of eclipse cycles per year for each of the LEO missions are within 15 percent of the other. One mission may see 100% sunlight, whereas the other will not. One is serviceable, the other is not. The two missions chosen do provide a good counterpoint to each other.

# 6.0 ANALYTICAL CHARACTERIZATION AND COMPARISON OF SOLAR DYNAMIC POWER CYCLES

The work reported in this section corresponds to the analytical study of the power systems which was the second activity of Task II plus Task X from the SOW.

## 6.1 OBJECTIVE

The size, weight, and performance for each solar dynamic cycle shall be determined. Each cycle shall be evaluated through a range of thermal storage peak temperatures to identify a realistic optimum peak cycle temperature taking into consideration pointing accuracy and concentrator errors. The effects of cycle temperature on weight, size, and efficiency shall be identified for each cycle and one category of power level. The selected peak temperatures for each cycle shall be used for the other power level category in the study.

Using the selected peak cycle temperature, the performance, mass, and size of the solar dynamic systems shall be determined for each mission. The output shall provide full performance parameters of efficiency, flow rates, pressure and temperature distribution, component and system weight and size including radiator parameters. This effort shall consider a solar dynamic system with a single collector.

A comparison shall be made between the systems included in this study with photovoltaic power systems for each selected power level. The comparison shall be with both state-of-the-art silicon cells (14.5% efficiency) and with advanced gallium arsenide cells with concentrators (22% efficiency). The comparisons shall be based on system efficiency, size, weight, reliability, etc. Advantages and disadvantages of the solar dynamic power systems versus photovoltaic systems shall be identified.

The study in this task shall be based on state-of-the-art receiver, concentrator, and power conversion technology to establish a technology base. The Contractor shall interface with the developers of Brayton, Rankine, and

Stirling cycle engines for required data relating to these systems.

## 6.2 TECHNOLOGY BASE

The purpose of the study was to identify technology areas that require research and advanced development. Therefore, the characterization of the solar dynamic and photovoltaic power systems had to be based on actual or expected near-term SOA. However, each of the power systems have evolved to a different level of maturity and confidence as to SOA performance, weight, etc. For example, open-cycle and closed-cycle Brayton experience is more extensive than either the free piston Stirling cycle or the liquid-metal Rankine cycle, and experience with planar silicon arrays is more extensive than GaAs concentrator arrays. The following discussion describes the SOA assumptions (and the level of technological maturity) as were used for this study for various power system components and subsystems.

### 6.2.1 Solar Dynamic Cycles State-of-the-Art

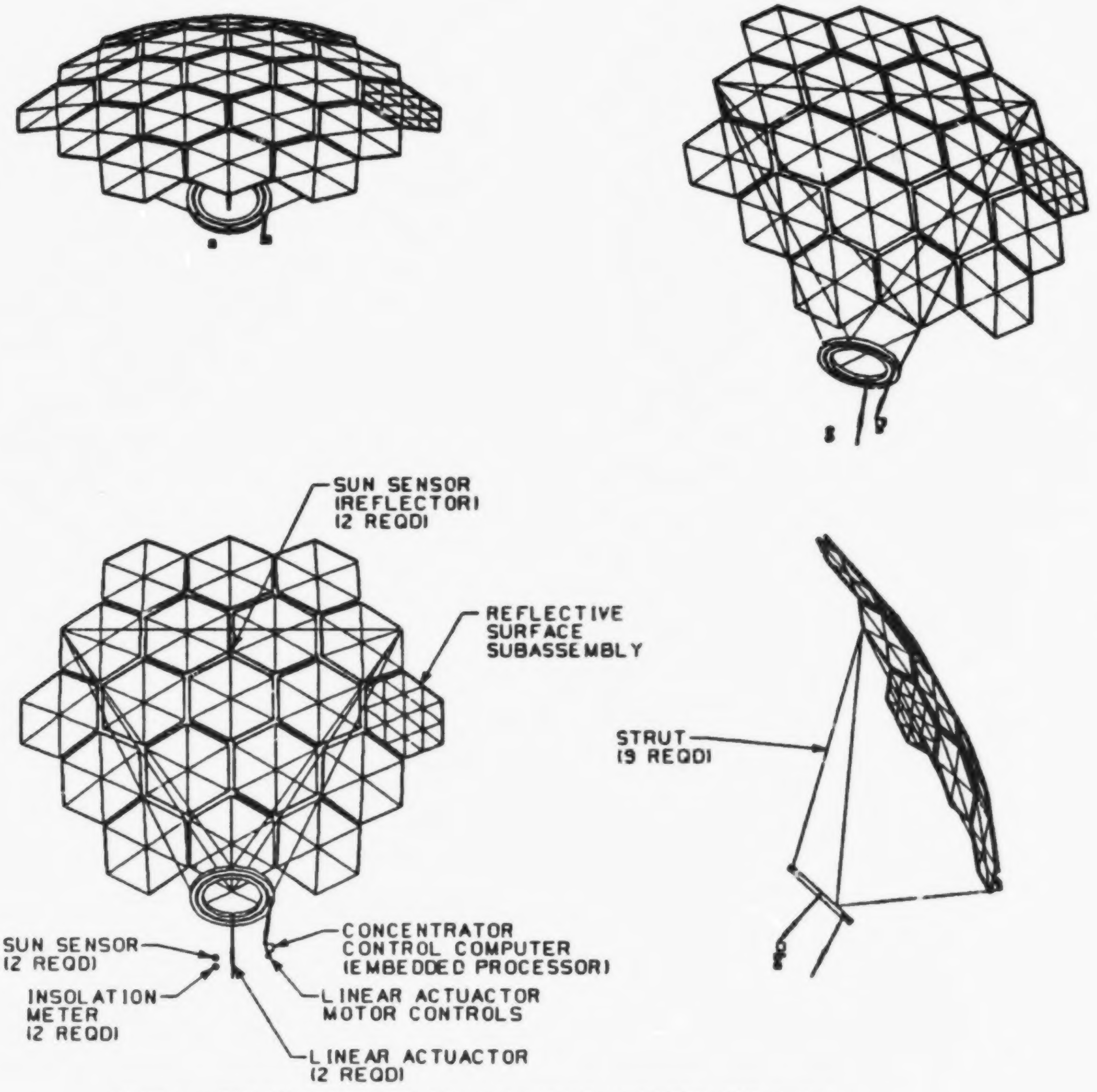
The current work on the Phase I and II Space Station power subsystems represents actual or near-term SOA technology, and this study drew substantially from the Space Station work. The main thrust of the study was to compare the three different solar dynamic power conversion cycles; therefore, trade studies which would tend to have a similar effect upon each cycle were not performed. As a result, the same type of concentrator and same type of radiator were used for each cycle. The method of heat transport to the engine tended to make receiver designs unique to the type of power conversion cycle. The Space Station reference design condition used for this study (for solar dynamic) was the December 1986 issue of Preliminary Analysis and Design Document DR-02 (ref. 1), hereinafter referred to as DR-02. This was the third and final issue of the Space Station WP-04 Power System Phase B Preliminary Analysis and Design document. Minor data refinements for solar dynamic as may have occurred during the WP-04 Phase C/D proposal were not included in this study, as that data was not publicly available.

### 6.2.1.1 Solar Concentrator

The concentrator selection was based on the work reported in the Solar Concentrator Advanced Development Program (ref. 3) performed by Harris Corporation in support of the Space Station. Three concentrator concepts were considered: 1) truss hex, 2) splined radial panel (SRP), and 3) domed Fresnel. The Space Station CBC truss hex configuration is shown in figure 6.2.1.1-1 and the hex panel subassembly is shown in figure 6.2.1.1-2. The latter two concepts were based on the proven Harris antenna technology and utilize a light weight umbrella-like support structure (fig. 6.2.1.1-3), whereas the truss hex concept utilizes a heavier beam construction. In each case, the concentrator surface is segmented for reasons of launch packaging and deployment, and herein lay the difference in technological maturity. To meet the particular needs of the initial Space Station, the ranking by Harris found the truss hex concept to be superior. The other two concepts were ranked nearly equal and were judged to be sound configurations with unique features better suited for other applications where more of a premium might be placed on reduced concentrator weight and launch volume.

The truss hex configuration was chosen for this study in recognition of the concept maturity to be realized from the ongoing Space Station design work. One design difference for the truss hex may be necessary in that higher concentration ratios appropriate to higher operating temperatures may require smaller (and greater in number) reflective segments for the concentrator (see figure 2.4-1). The SRP concentrator configuration (fig. 6.2.1.1-3) was also examined in comparison to the truss hex concentrator (see section 6.4.4), as the SRP concentrator represented an installed weight of less than half that of the truss hex configuration.

Mass characteristics for the truss hex concentrator were taken from DR-02, without any adjustment for possible change in reflective segment size. Weight characteristics for the SRP concentrator were taken from reference 3, and adjusted for structure (struts), controls, etc. When examining the specific weight of the concentrator, several factors must be kept in mind:


SUN SENSOR
(REFLECTOR)
(2 REQD)
REFLECTIVE
SURFACE
SUBASSEMBLY
STRUT
(9 REQD)
SUN SENSOR
(2 REQD)
INSOLATION
METER
(2 REQD)
CONCENTRATOR
CONTROL COMPUTER
(EMBEDDED PROCESSOR)
LINEAR ACTUACTOR
MOTOR CONTROLS
LINEAR ACTUACTOR
(2 REQD)

NOTE: INTERFACE STRUCTURE AND WIRING HARNESS NOT SHOWN

Rockwell International Corporation
Rocketdyne Division
Canoga Park, California

CBC CONCENTRATOR ASSEMBLY
PRELIM LAYOUT

C 02602 7R070020

NONE

Figure 6.2.1.1-1. Space Station CBC Concentrator Reflector and Structure

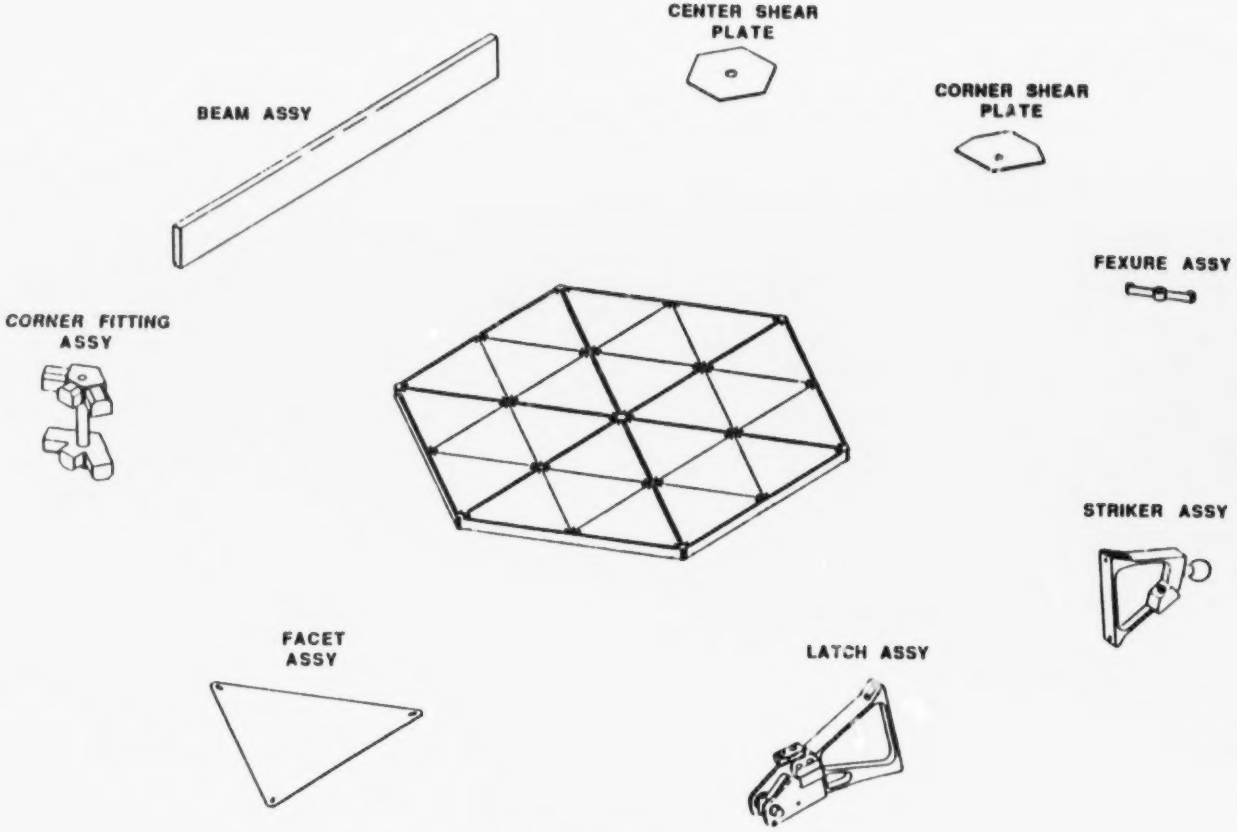

CENTER SHEAR PLATE
BEAM ASSY
CORNER SHEAR PLATE
FEXURE ASSY
CORNER FITTING ASSY
STRIKER ASSY
FACET ASSY
LATCH ASSY

Figure 6.2.1.1-2. Space Station Concentrator Reflector Panel Assembly

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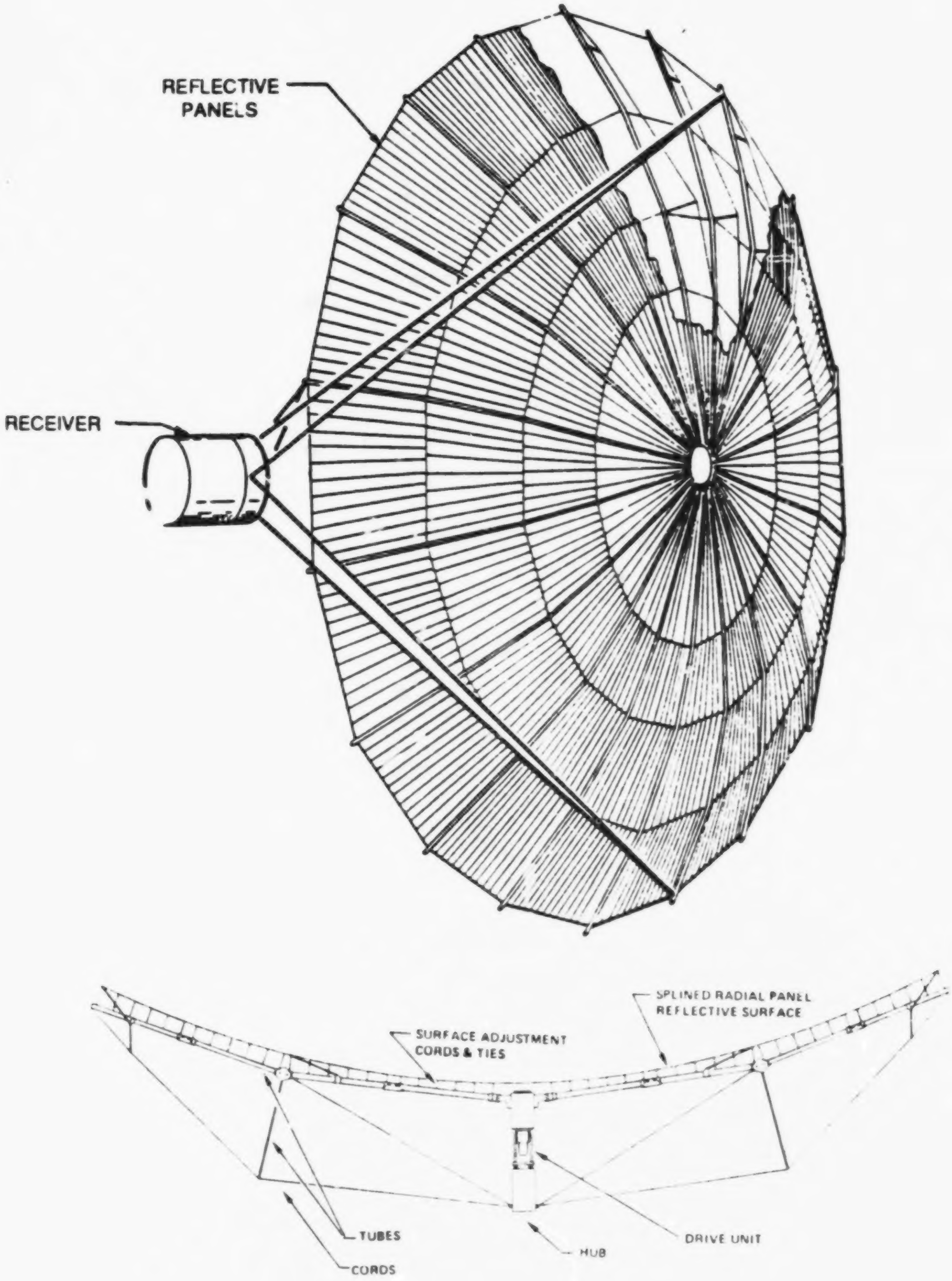
OF POOR QUALITY


REFLECTIVE PANELS
RECEIVER
SPLINED RADIAL PANEL REFLECTIVE SURFACE
SURFACE ADJUSTMENT CORDS & TIES
TUBES
CORDS
HUB
DRIVE UNIT

SECTION VIEW

Figure 6.2.1.1-3. Splined Radial Panel Concentrator

1. The truss hex concentrator has been configured in an offset design arrangement (refer to figure 2.4-1); therefore, concentrator aperture area is smaller than the concentrator projected area.
2. Structural shading plus gaps and overlaps existing between the triangular reflective segments result in useful reflective area being less than the area of the hex panels.
3. Concentrator specific weight may be defined simply as the reflective surface plus support structure, or may also include items such as support struts, controls, etc.

These factors and others were all considered in the study, thus resulting in increased estimates for concentrator specific weight as compared to the weight estimates for more preliminary and idealized concentrator designs.

### 6.2.1.2 Brayton Power Conversion Unit and Solar Receiver

The design used in this study for the Brayton power conversion unit (PCU) was closely patterned after the Brayton CBC proposed by Garrett Corporation for Space Station (fig. 6.2.1.2-1). The solar receiver design combined the Boeing Advanced Development (A/D) CBC receiver internals (TES and working fluid tubing, manifolds, etc.) as illustrated in figure 6.2.1.2-2, and the Garrett CBC receiver shell design with multi-foil insulation adjusted for higher temperature operation. The CBC cycle data from DR-02 (ref. 1) was used to calibrate both the CBC PCU and solar receiver computer codes, as well as the truss hex concentrator computer code, to the Space Station CBC operating conditions, before the codes were used for the conditions of this study. Higher temperature (1089K, 1500F) CBC operating data provided earlier by Garrett was also used for code calibration. Although these codes were developed independently of the Garrett work, the correlation of the various component algorithms to the detailed work of Space Station lends confidence to the quality of the study results at other design conditions.

The survey of peak operating temperatures for this study was to cover the range of 1100K to 1400K (1520F to 2060F). Implicit in this range of temperatures was the requirement for changes in materials of construction; however, no changes were made in the weight algorithms used for the PCU

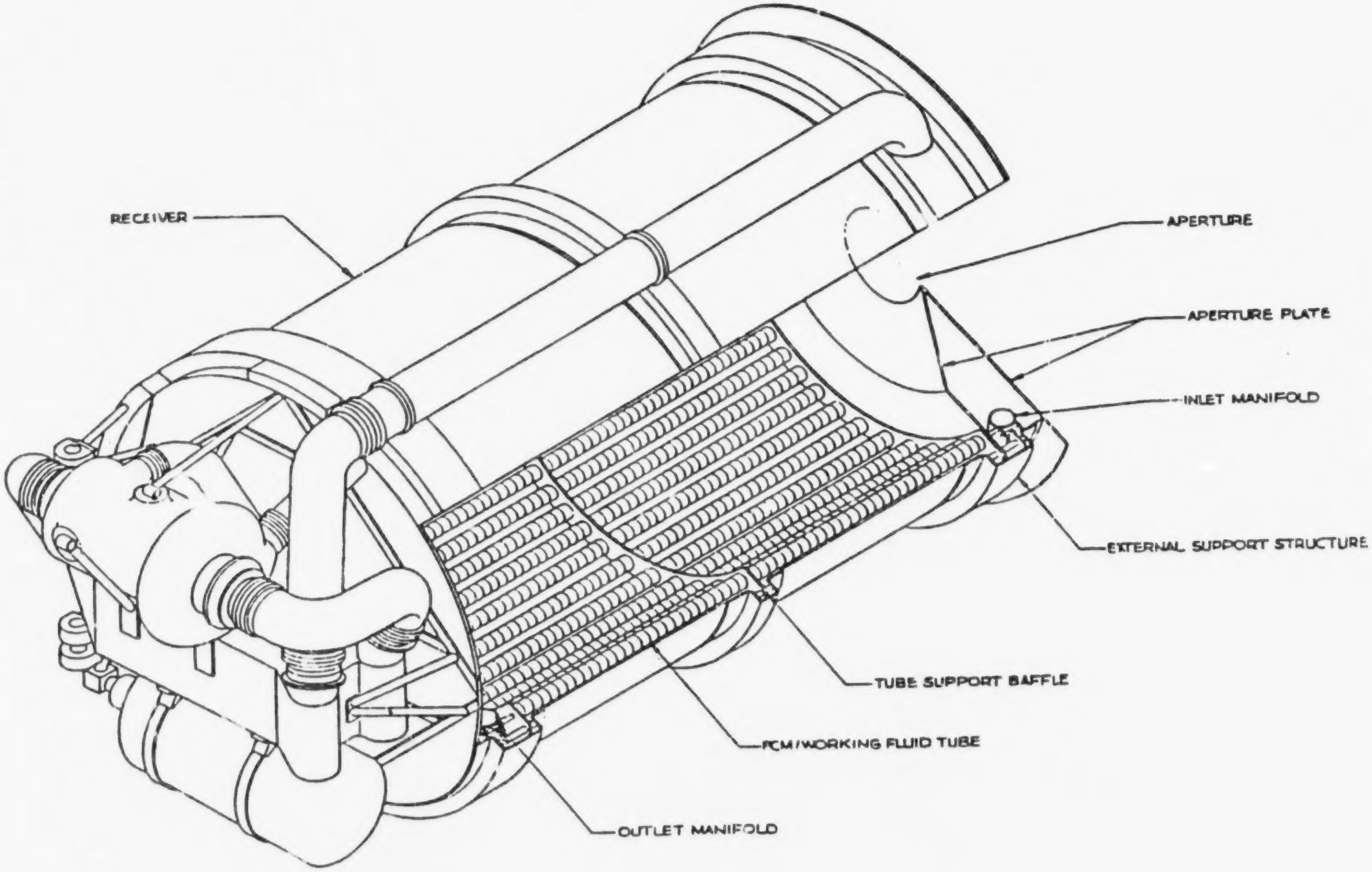

RECEIVER
APERTURE
APERTURE PLATE
INLET MANIFOLD
EXTERNAL SUPPORT STRUCTURE
TUBE SUPPORT BAFFLE
PCM/WORKING FLUID TUBE
OUTLET MANIFOLD

Figure 6.2.1.2-1. Space Station CBC Integrated Solar Receiver/PCU

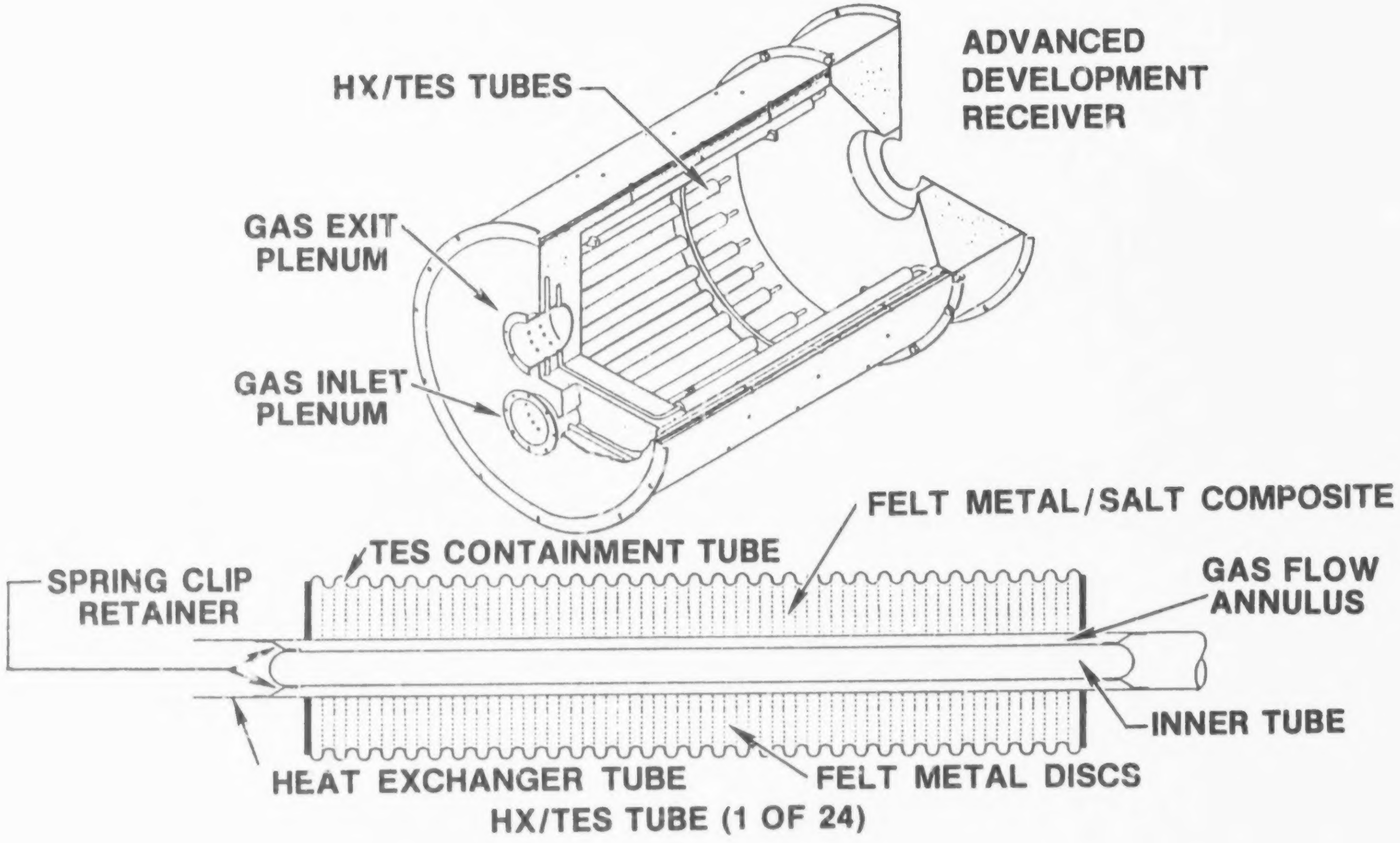

ADVANCED DEVELOPMENT RECEIVER
HX/TES TUBES
GAS EXIT PLENUM
GAS INLET PLENUM
FELT METAL/SALT COMPOSITE
TES CONTAINMENT TUBE
SPRING CLIP RETAINER
GAS FLOW ANNULUS
INNER TUBE
HEAT EXCHANGER TUBE
FELT METAL DISCS
HX/TES TUBE (1 OF 24)

Figure 6.2.1.2-2. CBC Advanced Development Receiver

components to account for materials changes. This assumption was used since only part of the PCU components are exposed to the high temperature, and since the PCU constitutes only a small fraction of the total power system weight.

The differences in receiver/TES configuration to account for higher operating temperatures include:

1. Appropriate changes in properties (heat of fusion, density) for each TES material being considered, and adjustments made to account for changes in containment volume of the TES material.
2. Geometric adjustments to receiver length and diameter to account for change in TES material quantity. For the 35 kWe CBC design, the same tube diameter and tube spacing were used as for the CBC A/D receiver design; for the 7 kWe design, the diameter and spacing were reduced.
3. Receiver aperture area (i.e. solar concentration ratio) was varied as a trade study in conjunction with concentrator surface accuracy and pointing error.
4. Receiver wall thickness (number of radiation shields) was varied to maintain approximately the same conduction loss heat flux (kW/m$^2$). Nickel foil proposed for Space Station was replaced with molybdenum foil.
5. The graphite aperture shield thickness was increased in proportion to receiver shell thickness for higher temperature TES cases (with higher concentration ratios).

A single computer code was developed to characterize the size and weight of the receiver and TES; that is, determining TES material quantity and TES containment sizing, receiver geometry and wall thickness, thermal losses, and aperture size. This code was used not only for the Brayton configuration, but the Rankine and both Stirling receiver/TES configurations as well through variation of input data to the code.

### 6.2.1.3 Rankine Power Conversion Unit and Solar Receiver

The Rankine cycle PCU characterization for this study was generally patterned after the Sundstrand Corporation 15 kWe ASTEC Program work (ref. 4), which was performed in the early 1960s. The ASTEC PCU was a Rankine cycle designed for high temperature liquid metal operation. The receiver/TES configuration for this study was generally patterned after the Sundstrand organic Rankine receiver proposed for Space Station, although such a design would have to be substantially reconfigured for two-phase operation and possible reheats. Further design detail was not pursued as the Rankine cycle was subsequently eliminated from consideration.

The Rankine cycle characterization considered peak operating temperatures above 1400K (up to 1543K (2318F) for $MgF_2$ TES) in order to achieve higher cycle efficiencies. Besides peak temperature, additional tradeoffs were examined to achieve higher cycle efficiencies, including choice of liquid metal working fluid (K, Na) and inclusion of reheats and multistage turbines in order to operate with higher pressure ratios.

The outcome of the tradeoffs resulted in Rankine cycle thermodynamic efficiencies ranging generally from 15% to 30% versus Brayton and Stirling cycle PCU efficiencies of about 35-40%. For example, figure 6.2.1.3-1 shows a T-S diagram from reference 4 for the ASTEC system, using rubidium and a single reheat expanding to 2% moisture, which resulted in an efficiency of only 25.2%. Higher Rankine cycle PCU efficiencies could have been achieved for very high peak temperatures ($\geq$1543K) with two or three reheats; however, this would be beyond near-term SOA. At peak temperatures of 1100K to 1250K, which were optimum (minimum system weight) for the Brayton and Stirling cycles, the Rankine cycle with a single reheat only achieved about a 25% efficiency.

For a solar dynamic power system, most of the weight accrues from the concentrator, receiver/TES, and radiator, and these weights are approximately proportional to the inverse of cycle efficiency. As a result, Rankine solar dynamic power system weights were not competitive with the other two cycles, and the Rankine cycle was dropped from further consideration. Accurate Rankine system weights were not developed, as such an effort was not warranted.

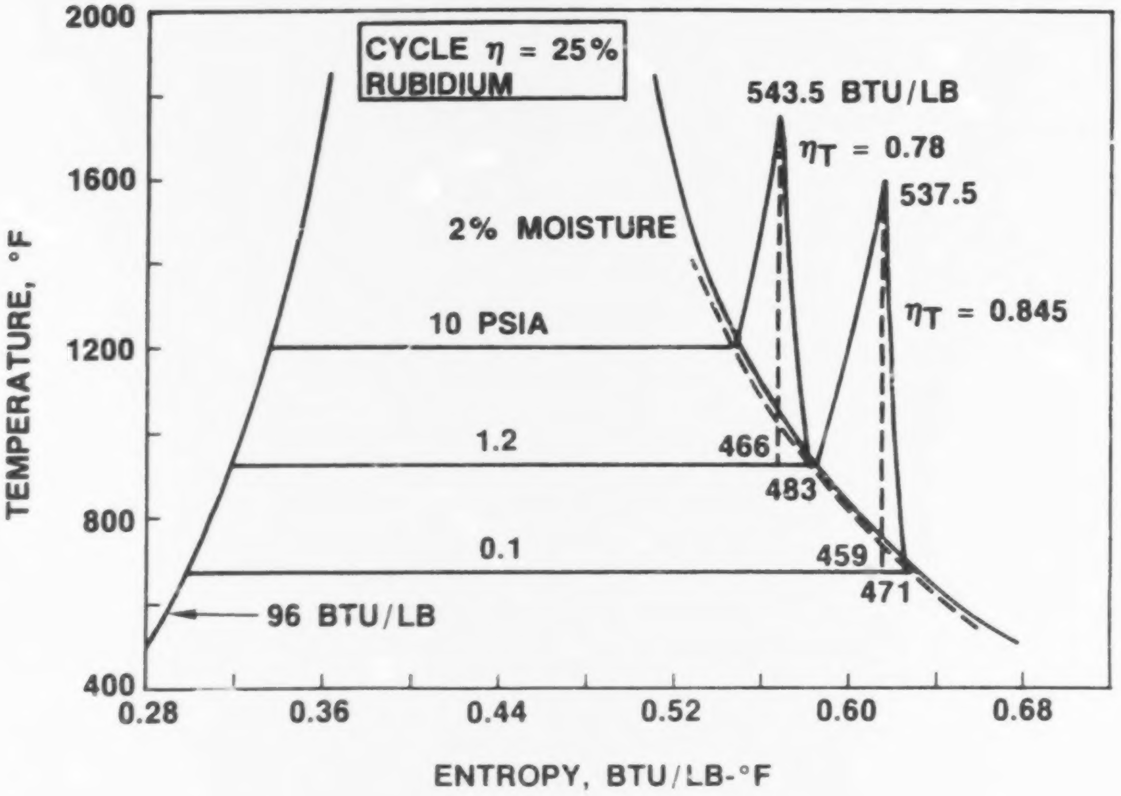


Figure 6.2.1.3-1. Rankine Thermodynamic Cycle with Single Reheat

### 6.2.1.4 Stirling Power Conversion Unit and Solar Receiver

Characterization of the Stirling cycle PCU is handled somewhat differently than the Brayton and Rankine cycles. The latter are basically turbine expanders coupled to rotating alternators, with necessary compressors or pumps and heat exchangers. All of those components have extensive working experience and analytical modeling characterizations. The free piston Stirling engine (FPSE), a fairly recent invention, is a sealed engine so that heat is transferred to and from the engine working fluid through heat exchangers, and the developed power is provided through linear motion, which may be coupled to a linear alternator sealed within the engine or the power may be used to drive an external hydraulic motor. Analytical characterization of this engine involves very detailed non-steady thermal analysis.

FPSE engine designers have developed overall engine characterizations for use by power systems designers which graphically interrelate engine efficiency and engine specific weight with engine temperature ratio. The efficiency, expressed as fraction of Carnot efficiency, includes the linear alternator efficiency, and engine specific weight includes alternator weight as well. The 25 kWe (2 x 12.5) Stirling Space Power Demonstrator Engine (SPDE) and the new 25 kWe Space Stirling Engine (SSE) have both been designed for temperature ratio ($T_H/T_C$) of 2.0, a ratio more suitable for nuclear power conversion application. Trade studies for solar powered Stirling engines have usually resulted in optimum $T_H/T_C$ in the range of 2.5 to 3.0. A result of the increased temperature ratio (reduced $T_C$) is a marked increase in engine power output for fixed thermal input, due to increased thermal conversion efficiency (e.g. 25 kWe to 34-39 kWe).

The Stirling cycle weight and performance characterizations for this study were obtained from two sources. The first was under a subcontract to Sunpower, Inc. (ref. 5). The FPSE data was patterned after both the two-cylinder SPDE, which has been built and tested, and the new single-cylinder 25 kWe SSE (fig. 6.2.1.4-1). Both Sunpower and Mechanical Technology Incorporated (MTI) have contributed to the design of the NASA-LeRC sponsored SSE.

The subcontract for FPSE data was performed by Gedeon, Associates for Sunpower. The Gedeon work extrapolated engine design and performance parameters from the 25 kWe SSE design to the requested power levels of 8 kWe and 40 kWe gross power output from the alternator. These extrapolations were done for $T_H/T_C$ of 2.0; therefore, at solar temperature ratios, the engine designs would result in considerably higher power levels. As a result, the 35 kWe net power (~40 kWe gross) design for this study was closer to the 25 kWe SSE design than to the Gedeon 40 kWe design, based on thermal power input to the engine.

The Gedeon work (ref. 5) found that the engine efficiency value at the 40 kWe power level, as predicted by the extrapolation method, fell below that of the 25 kWe SSE engine design. The NASA-LeRC Project Office and Sunpower recommended that the same values be used for both the 25 kWe and 40 kWe

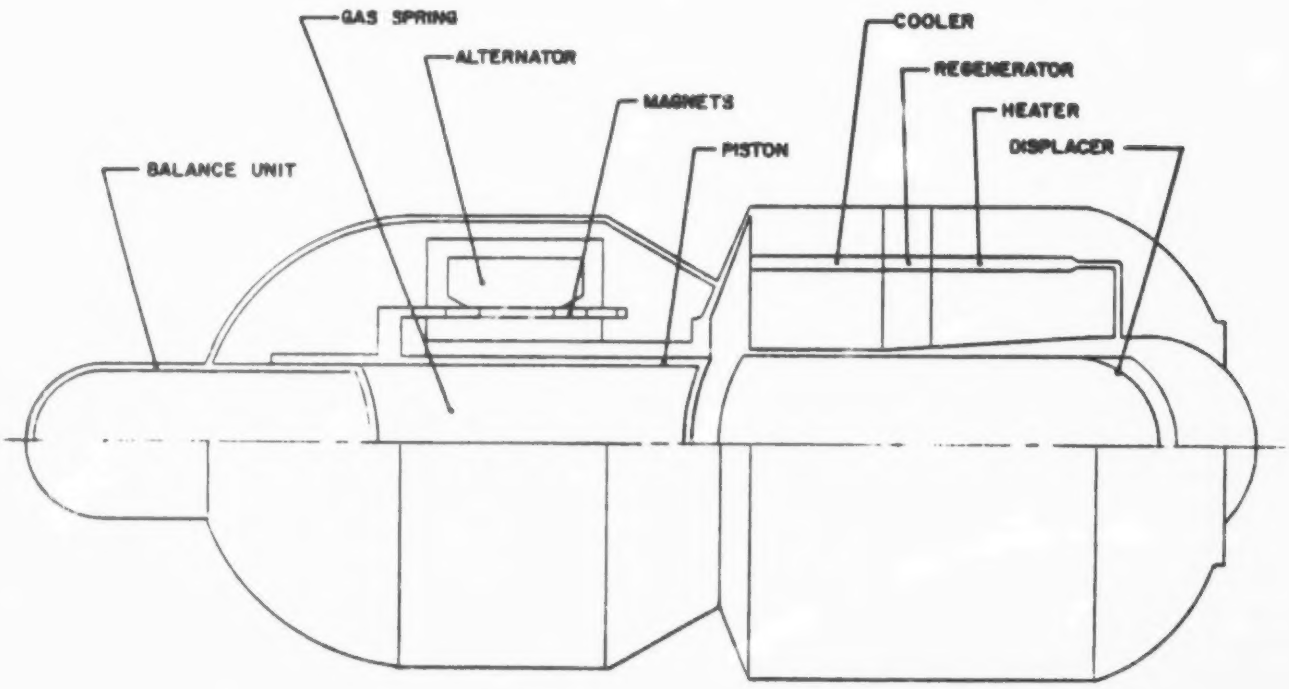

GAS SPRING
ALTERNATOR
MAGNETS
PISTON
BALANCE UNIT
COOLER
REGENERATOR
HEATER
DISPLACER

Figure 6.2.1.4-1. Sunpower 25 kWe Space Stirling Engine (SSE)

engines (58.7% of Carnot efficiency including alternator efficiency, at a specific weight of 5.5 kg/kWe for $T_H/T_C$ of 2.0). The efficiency and specific weight figures from reference 5 were used for the 8 kWe design (57.2% of Carnot at 7.0 kg/kWe for $T_H/T_C$ of 2.0).

The second source of information for FPSE characterization was information provided by MTI (a combination of Sunpower data up to a temperature ratio of 2.0 and an extrapolation by MTI to a temperature ratio of 3.0), which is reproduced in figure 6.2.1.4-2. In the final analysis, this was the data used to characterize the 35 kWe design engine, providing the necessary interrelationship between engine efficiency and weight for a range of temperature ratios.

In the design of a FPSE, a tradeoff exists to design for higher engine efficiency at the expense of increased engine weight. Engine weight is less than 10% of power system weight, but engine efficiency directly effects the size and weight of the other major components: the concentrator, receiver/TES, and radiator.

Maturity of the SOA for the FPSE performance and design conditions is less than for Brayton, as fewer engine designs and hours of operation exist for the FPSE. Performance predictions for the FPSE are believed to be achievable, but have not yet been demonstrated, whereas Brayton engine performance conditions have more nearly been demonstrated. Both MTI and Sunpower have detailed FPSE design and performance codes which correlate well with one another and with engine test data at the higher temperature ratios for a solar Stirling.

Two Stirling engine heater head designs were considered, one based on a pumped liquid metal heat transport approach, and the other based on heat pipes for heat transport. The receiver/TES configuration for the pumped loop approach was patterned after the Rocketdyne Receiver/Thermal Storage Assembly Development Project tested in 1986 in support of the Space Station design (fig. 6.2.1.4-3). The second configuration, the heat pipe receiver/TES design was developed for this application to match with the SSE heat pipe heater head configuration (fig. 6.2.1.4-4).

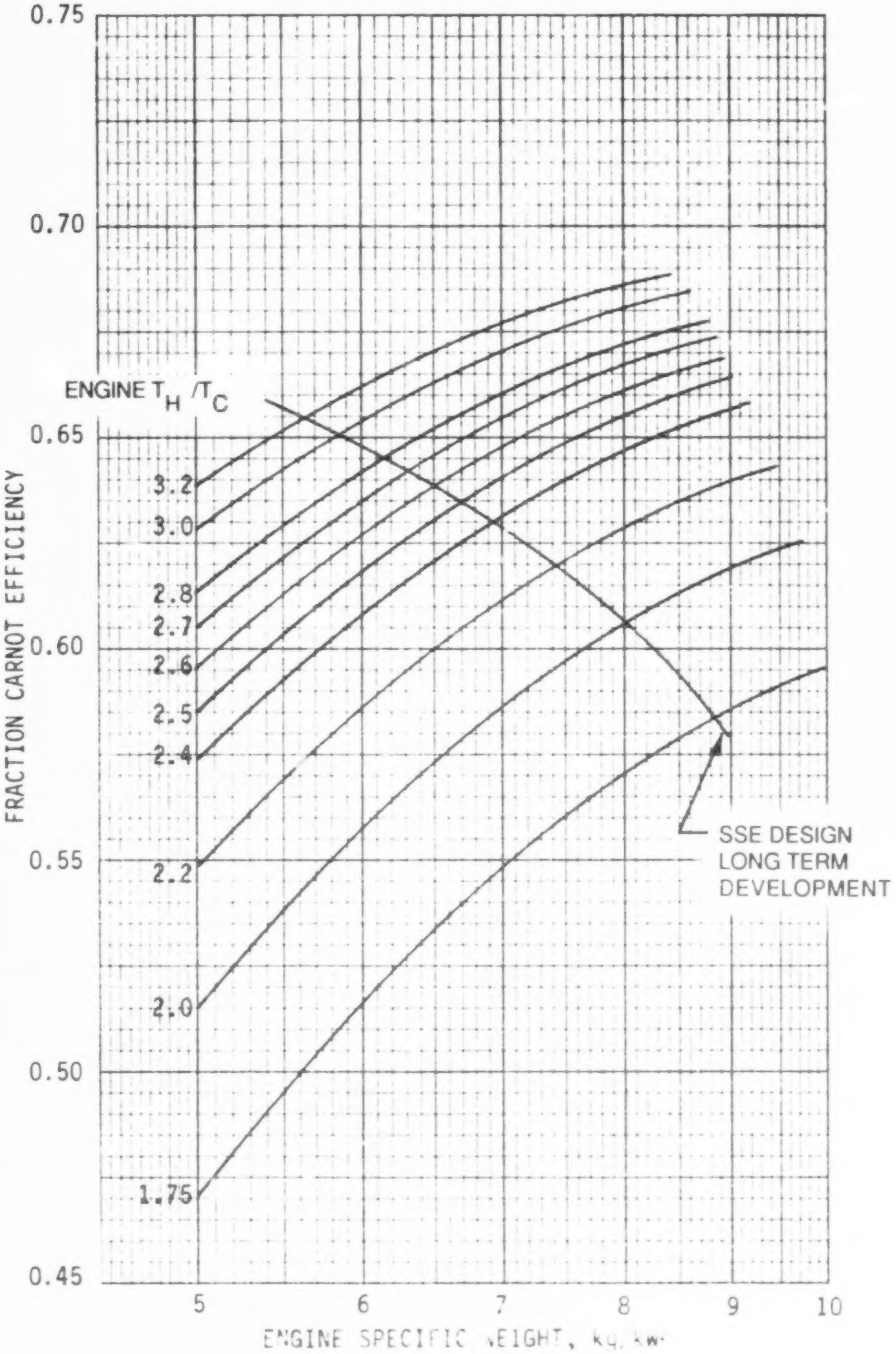


Figure 6.2.1.4-2. Stirling Engine Efficiency Performance Map

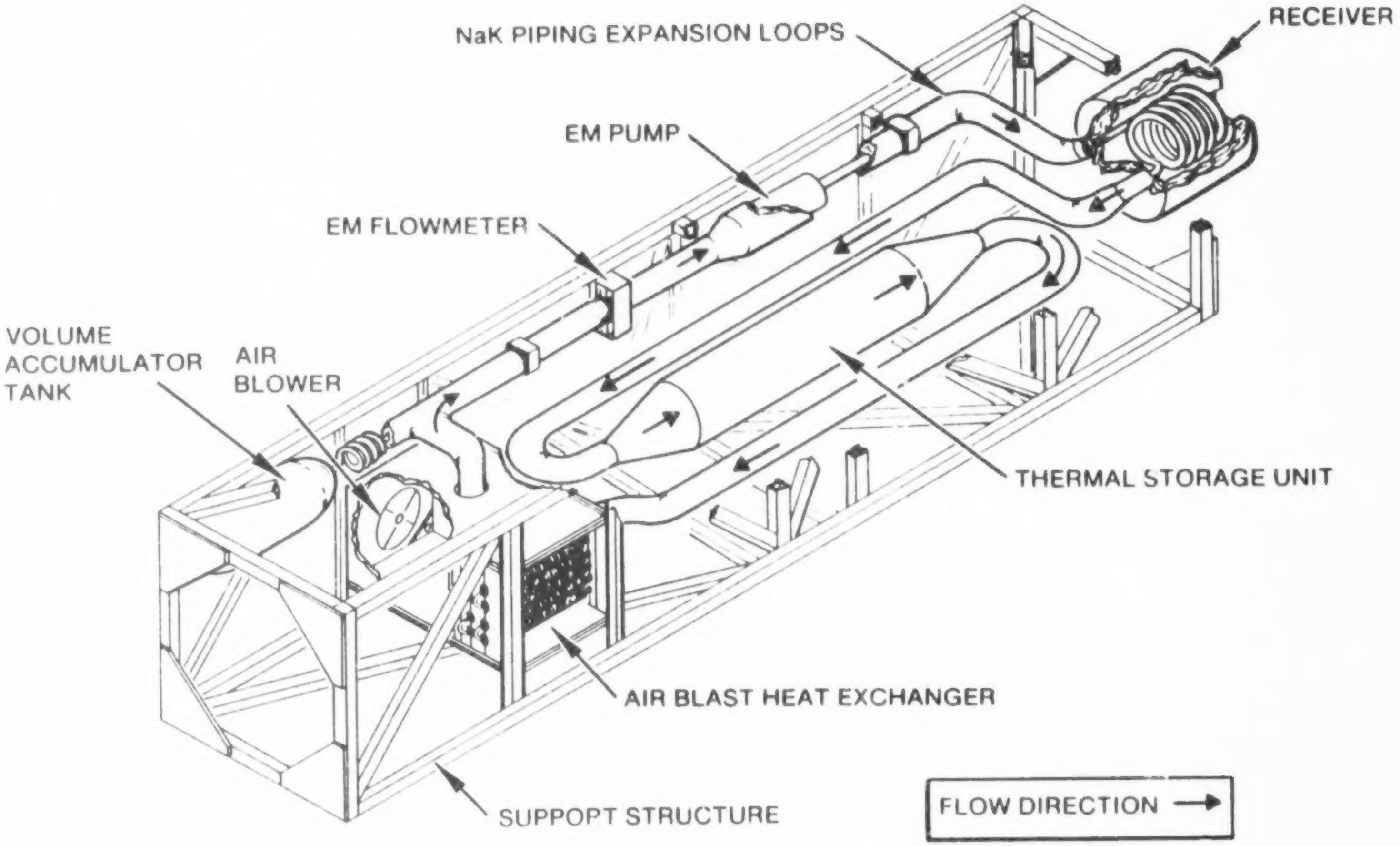

RECEIVER
NaK PIPING EXPANSION LOOPS
EM PUMP
EM FLOWMETER
VOLUME ACCUMULATOR TANK
AIR BLOWER
THERMAL STORAGE UNIT
AIR BLAST HEAT EXCHANGER
SUPPORT STRUCTURE
FLOW DIRECTION →

Figure 6.2.1.4-3. Pumped Loop Receiver Concept

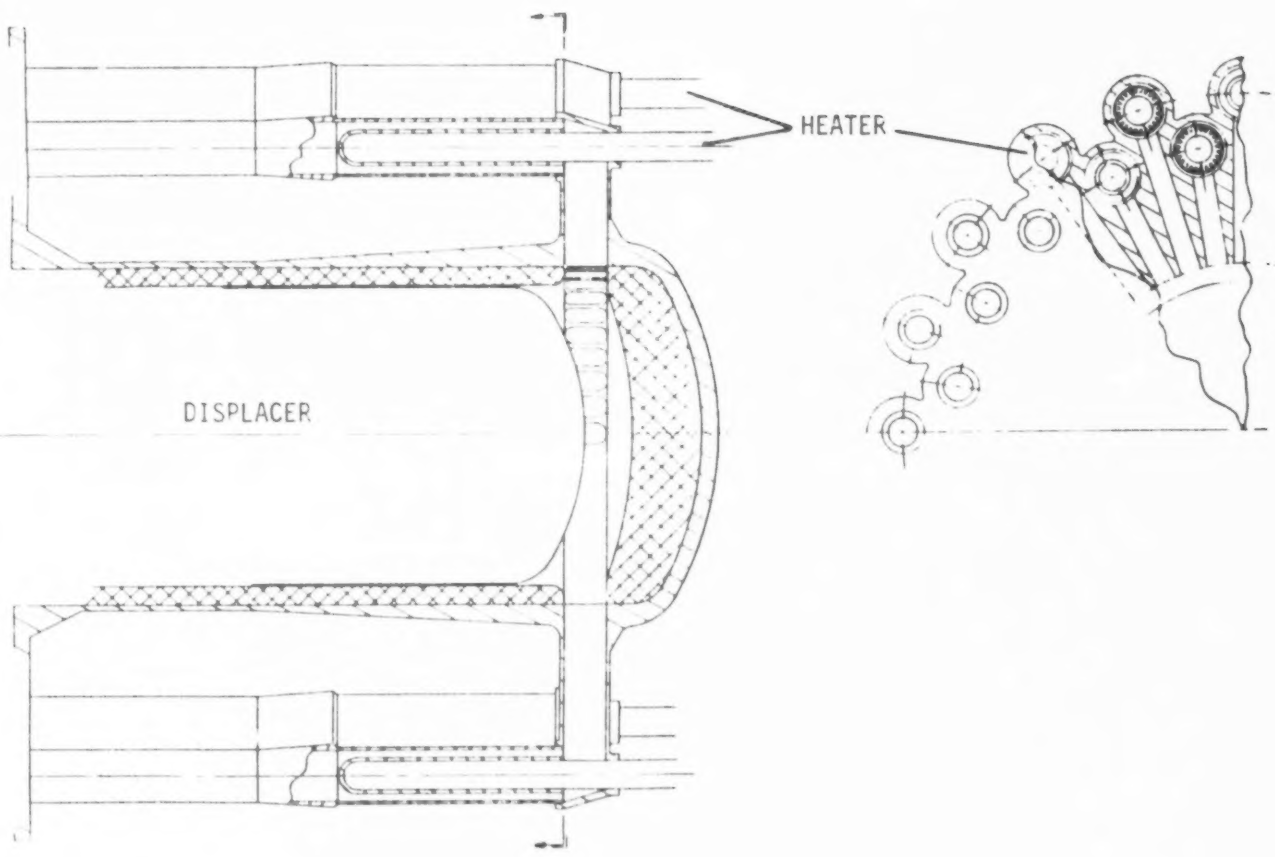

HEATER
DISPLACER

Figure 6.2.1.4-4. Stirling Engine Heat Pipe Heater Configuration

As was the case for the Brayton cycle, no changes were made in the weight algorithms for either the Stirling engine or the receiver to account for changes in materials of construction due to higher operating temperatures. Of course, changes due to TES material, geometry, and insulation thickness were considered.

The two different receiver/TES design concepts were developed corresponding to the different Stirling engine heater head design approaches. The pumped loop design, illustrated in figure 6.2.1.4-5, employs a small receiver with a simple tubular helical coil to absorb the solar energy. The heated liquid metal then passes through the remote TES vessel similar in construction to a shell and tube heat exchanger but containing TES cannisters in place of the tubes. The liquid metal is pumped through the engine and then to the receiver inlet. This design results in a small receiver, a remote TES design which contributes to a significant liquid metal inventory, requires an electromagnetic (EM) pump which has low efficiency, and is a design which would contribute to single point failure concerns.

The heat pipe Stirling receiver/TES design concept employs heat pipes for both heat absorption (the primary heat pipe), and to interface the TES and the engine (the secondary heat pipe). The 35 kWe and 7 kWe versions of this design concept are shown in figures 2.4-2 and 6.2.1.4-6. The arrangement of the heat pipes and TES material is shown in figure 6.2.1.4-7, sized for the 35 kWe engine (1 of 40 total).

Although a single heat pipe from the receiver to the engine heater would appear to be functionally satisfactory, the large volume shrinkage of the TES material upon freezing must be dealt with. The dual heat pipe design ensures that the void formed in the TES material upon freezing during eclipse will be next to the tube wall that will be heated during the next solar input period. This design results in a small receiver in that the TES is external to the receiver cavity. The dual heat pipe arrangement for TES material containment and heat transfer results in significant containment weight in comparison to the weight of the TES material proper. Heat pipes are very efficient thermal transport devices. The large number of heat pipes employed in the receiver avoids single point failure concerns, in that failure of a single heat pipe

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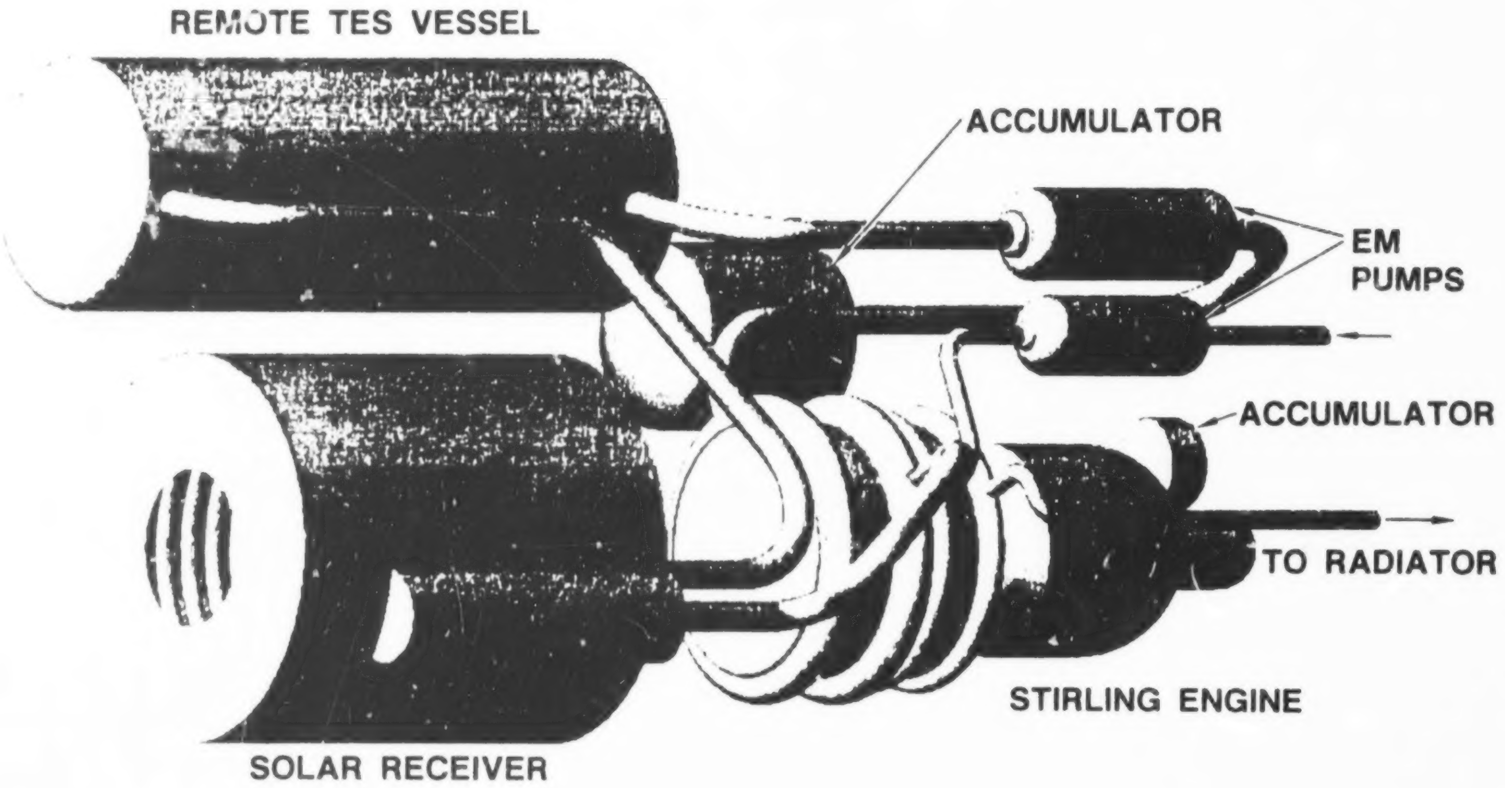
REMOTE TES VESSEL
ACCUMULATOR
EM
PUMPS
ACCUMULATOR
TO RADIATOR
STIRLING ENGINE
SOLAR RECEIVER

Figure 6.2.1.4-5. 35 kWe Stirling Engine/Pumped Loop Receiver

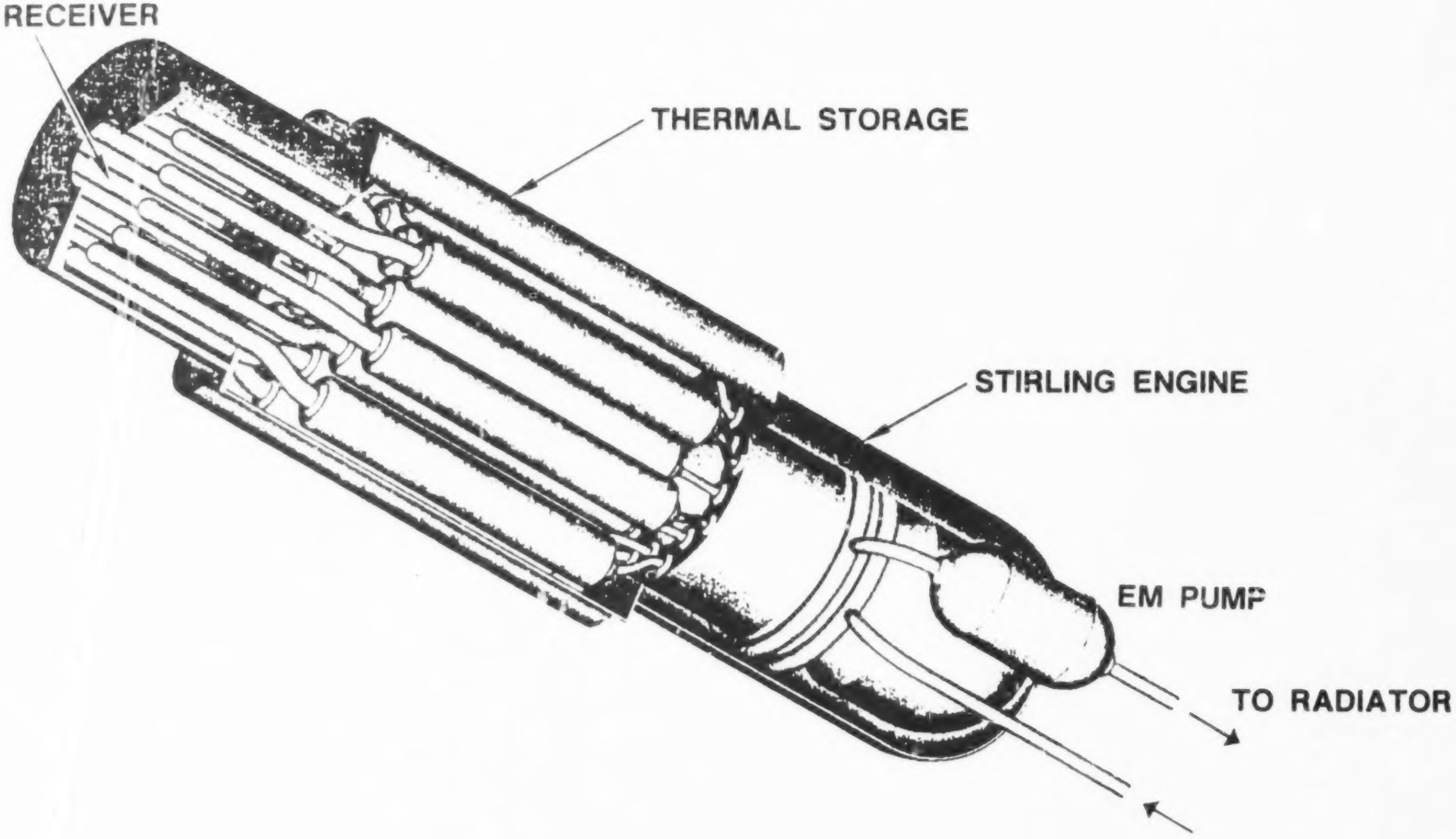
RECEIVER
THERMAL STORAGE
STIRLING ENGINE
EM PUMP
TO RADIATOR

Figure 6.2.1.4-6. 7 kWe Heat Pipe Stirling Engine/Solar Receiver

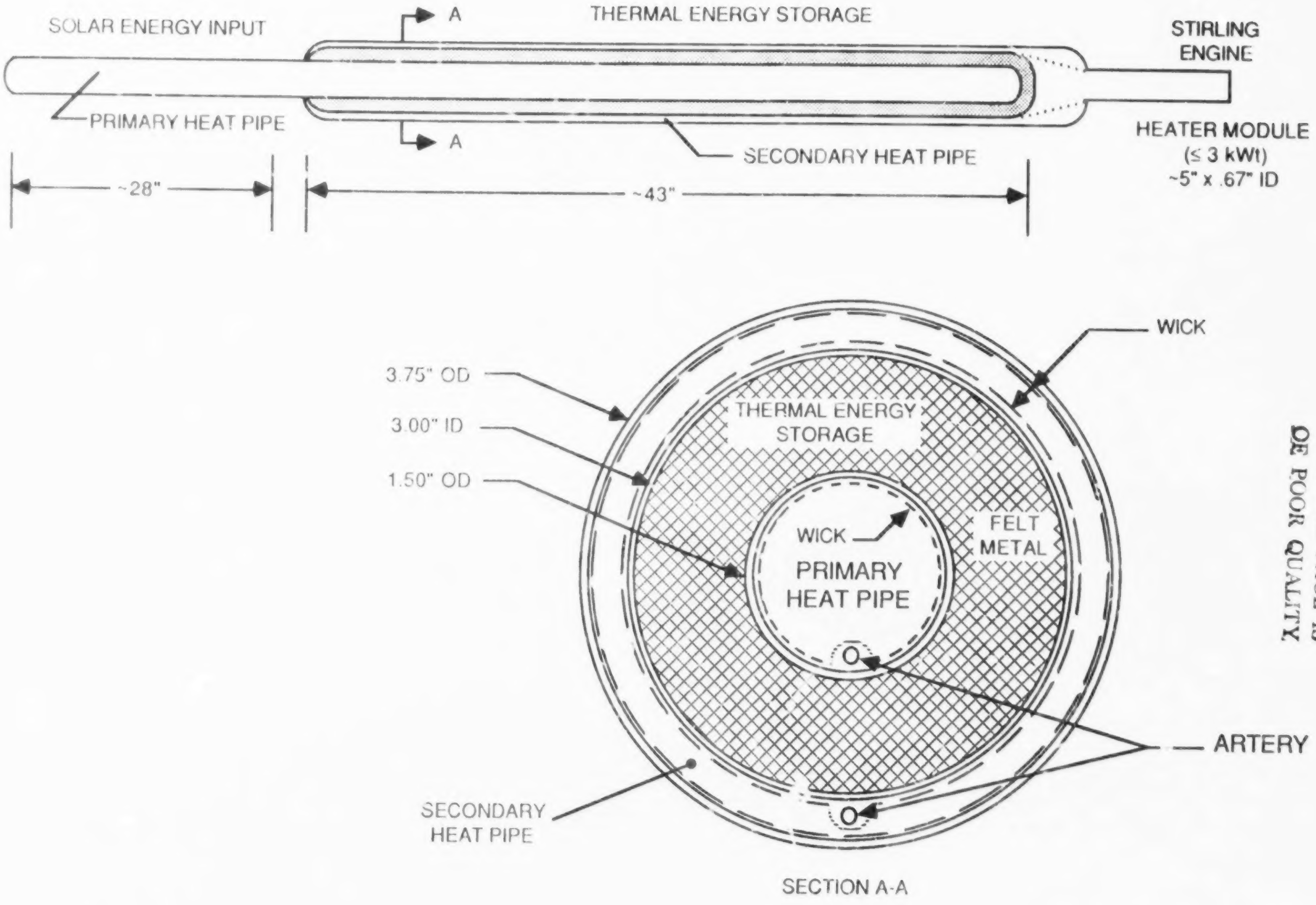
SOLAR ENERGY INPUT
A
THERMAL ENERGY STORAGE
STIRLING ENGINE
PRIMARY HEAT PIPE
A
SECONDARY HEAT PIPE
HEATER MODULE
(≤ 3 kWt)
~5" x .67" ID
~28"
~43"
WICK
3.75" OD
3.00" ID
1.50" OD
THERMAL ENERGY STORAGE
WICK
PRIMARY HEAT PIPE
FELT METAL
ARTERY
SECONDARY HEAT PIPE
SECTION A-A

Figure 6.2.1.4-7. Stirling Engine Heat Pipe Energy Storage Module Design

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would normally allow continued engine operation at some loss in power.

The liquid metal used for heat transport for either the pumped loop receiver or waste heat exchange loop would be a eutectic mixture of sodium (Na) and potassium (K) referred to as NaK78 or NaK, with 77.7% by weight potassium. Melting point is 280K (12F). Vapor pressure for the hot loop with LiF TES would be 1-2 atmospheres. For higher temperature TES materials, sodium was substituted for NaK in the hot loop. Sodium is a superior heat transfer fluid to NaK, with somewhat higher density and much higher heat capacity, but it has a distinct disadvantage in that the freeze point is 371K (208F), so trace heating would be required for periods of non-operation. NaK is used in the cold heat transport loop for both the pumped loop Stirling and heat pipe Stirling configurations. The use and handling of NaK and sodium are present-day SOA through years of experience gained in the nuclear power field.

### 6.2.1.5 Electromagnetic Pumps

Electromagnetic (EM) pumps were chosen for the Stirling cycle liquid metal heat transport loops. Of the several configurations considered (ALIP - annular linear induction EM pump; FLIP - flat linear induction EM pump; helical induction EM pump), the ALIP configuration shown in figure 6.2.1.5-1 was found to be best suited considering pump head and flow requirements, efficiency, and weight. Such a pump (nominal 25 gpm, 25 psi head, NaK at 1400F designed and built by MHD Systems, Inc.) was used quite successfully on the Rocketdyne subscale solar receiver shown in figure 6.2.1.4-3. Two similar ALIP EM pumps were used as the basis for this study (H.E. Adkins, 1986, MHD Systems, Inc., Kennewick, WA, private communication):

1. 1600F design temperature, sodium, 101 gpm, 12 psi head, 12.2% efficiency, 180 V ac 3-phase power, 110 lb weight.
2. A paper design for a 15 gpm pump with estimated efficiency of 4-5% and 50 lb weight.
3. These two points were used to estimate pump operation at the intermediate flow conditions. Efficiency for high temperature operation was increased by a factor of 1.2 to account for improvement due to pump design optimizations.

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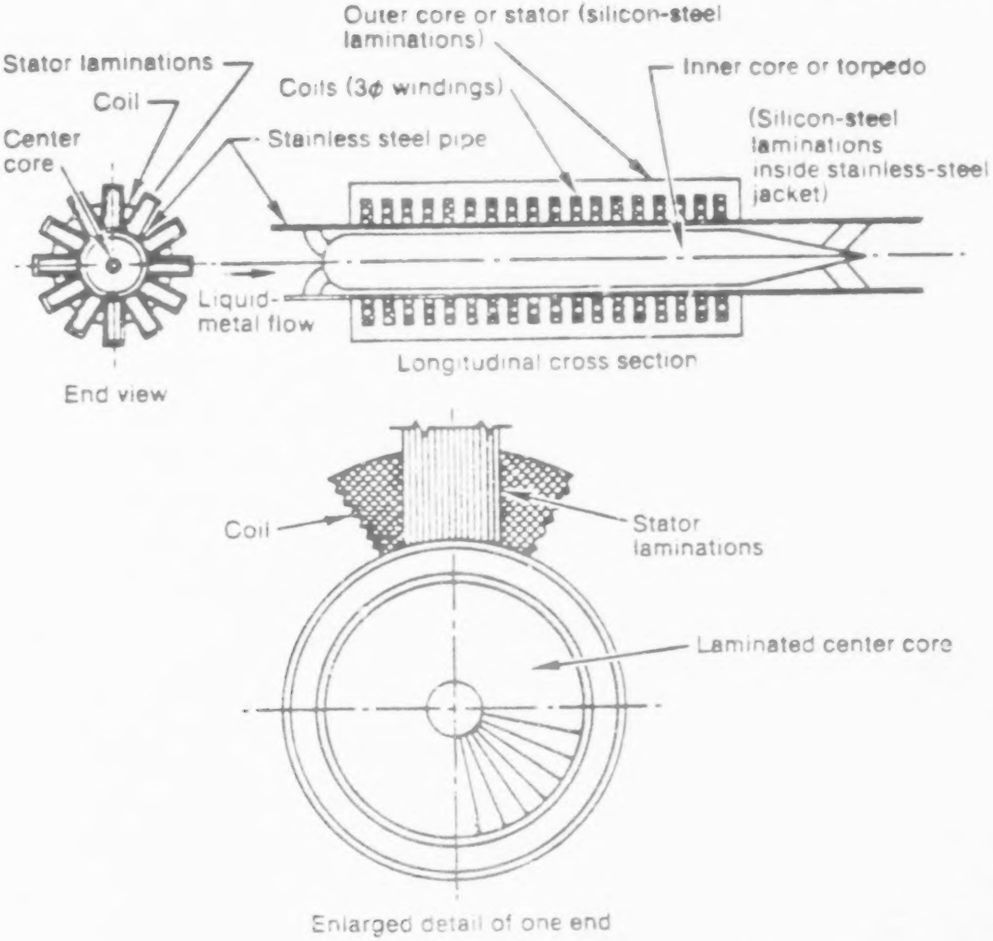

Outer core or stator (silicon-steel laminations)
Stator laminations
Coil
Center core
Coils (3ϕ windings)
Inner core or torpedo
Stainless steel pipe
(Silicon-steel laminations inside stainless-steel jacket)
Liquid-metal flow
Longitudinal cross section
End view
Coil
Stator laminations
Laminated center core
Enlarged detail of one end

Figure 6.2.1.5-1. Cross Section of ALIP EM Pump

4. At the low operating temperatures of the Stirling waste heat transport loop, pump efficiencies would be even higher and weight would be slightly reduced.

Pump sizes for the Stirling application are relatively small as relates to past experience with EM pumps. At 35 kWe, flows would be 62 gpm and 28 gpm (hot and cold loops), whereas at 7 kWe, flows would be 15 gpm and 11 gpm. The natural trend is for a reduction in pump efficiency with reduction in design flowrate, as the losses in the pumping process cannot be reduced proportionally. Further analytical studies and construction and test of a low-flowrate, flightweight (10-15 gpm) ALIP pump would be necessary to establish confidence in small pump SOA. The limited design experience at these low flowrates, plus the absence of any detailed hydraulic analysis of the pressure drops of the liquid metal pumped loops, resulted in a less than desired confidence in the predicted pump power requirements.

### 6.2.1.6 Waste Heat Radiator

The radiator selection was based on the heat pipe radiator R&D technology study (ref. 2) performed by Grumman Space Systems Division in support of the Space Station. The heat pipe radiator configuration was chosen for this study, rather than a pumped loop configuration, primarily due to one of the selected missions being unserviceable. For each cycle design, engine heat rejection was through an intermediate heat exchanger to a secondary pumped fluid loop to the radiator. The secondary loop arrangement would allow ready substitution of a pumped loop radiator for the serviceable type of mission, thereby resulting in a weight savings for each power cycle. Multiple redundant pumped loops would be necessary for purposes of reliability; therefore, pumped loop radiators become larger and heavier for unserviceable missions due to the number of redundant loops required, and ordinarily are not selected for such missions.

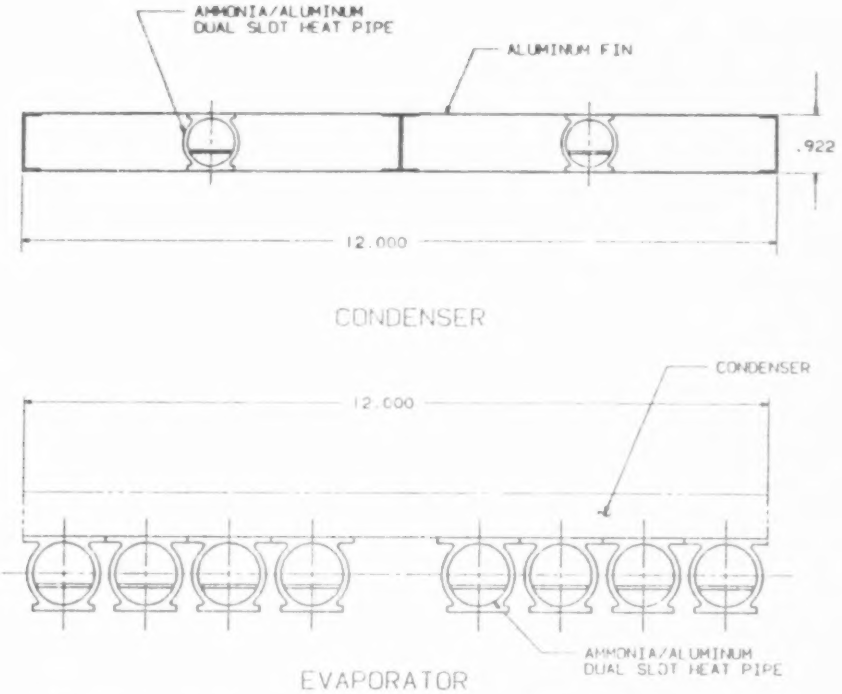
The Grumman work considered the following heat pipe combinations: aluminum/ammonia, stainless steel/methanol, and titanium/methanol. For this study, peak operating temperatures for the working fluids were limited to less than 350K (170F) for ammonia based on vapor pressure, and less than ~400K (260F) for methanol based on decomposition concerns. Operating temperatures for both the CBC and Stirling cycles were such that the titanium/methanol heat pipes were suitable for both. The CBC cycle radiator would operate over a range of inlet-to-outlet temperatures such that a hybrid radiator could be considered, comprised of aluminum/ammonia pipes for the lower temperatures and titanium/methanol for the higher temperatures. No weight advantage was realized, however, due to the aluminum and titanium pipes having nearly the same weight as a result of differing wall thickness for micrometeoroid and debris protection. The methanol heat pipes could be transport capacity limited at the lower temperatures, possibly requiring substitution of aluminum/ammonia heat pipe instead.

The heat pipe configuration chosen was the dual-slot design, as shown in figure 6.2.1.6-1, which results in lower weight than the mono-groove design. (Note: Figures 6.2.1.6-1 through 6.2.1.6-4 were all obtained from reference 2.) The heat pipe panel is comprised of a 0.305 m by 0.710 m (1 ft by 2 ft) evaporator section and a 0.305 m by 7.62 m (1 ft by 25 ft) condenser section for the titanium/methanol configuration. Development work conducted on the dual-slot design has established feasibility and preliminary performance to the point where it may be considered near-term SOA.

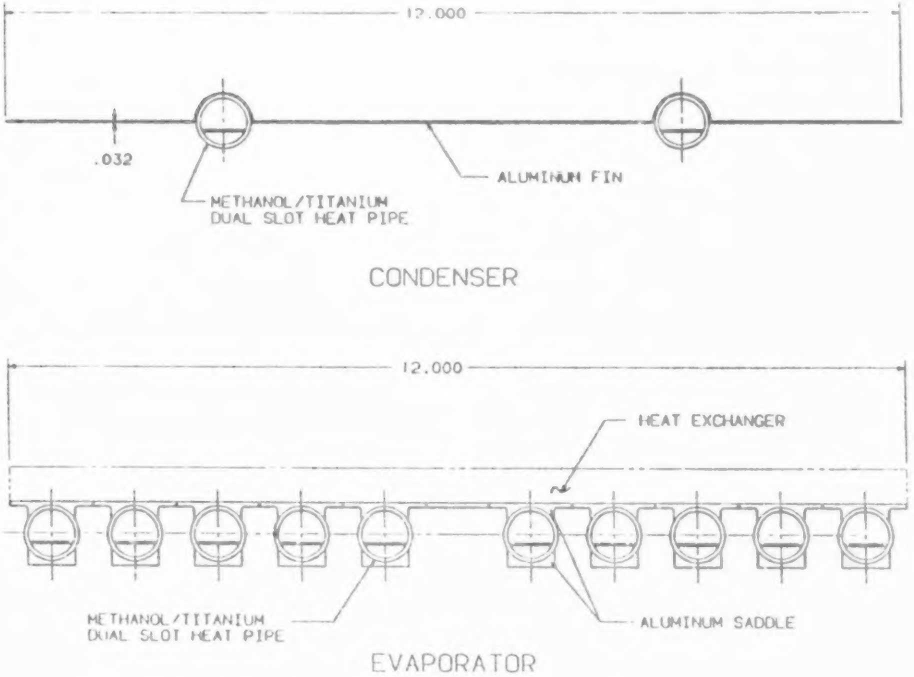
Attachment techniques for the heat exchanger boom and heat pipe evaporators have been established and some development work performed. The three approaches are: a whiffletree clamp (fig. 6.2.1.6-2), a quick disconnect (fig. 6.2.1.6-3), and a folding boom configuration (fig. 6.2.1.6-4). The whiffletree clamp weighs twice that of the quick disconnect; however, configuring the heat exchanger boom as doublesided (as shown in figure 2.4-1) distributes the whiffletree weight over two heat pipes, making whiffletree and quick disconnect weights equivalent. Weights for a folding design, which is similar to the quick disconnect design in that the heat pipe evaporators are bonded (probably brazed) to the heat exchanger boom, would result in a radiator weight similar to the other two configurations. The folding boom design could be preferable where remote deployment is required.

### 6.2.2 Photovoltaic Power Systems State-of-the-Art

The current or near-term SOA for performance of photovoltaic arrays was specified in the SOW (section 6.1) as solar cell efficiencies of 14.5% for planar silicon and 22% for GaAs concentrators. The silicon array design was patterned closely after the current Space Station detailed design work (which used a 12.9% cell efficiency). The GaAs concentrator array was patterned after GaAs cell designs by Hughes Aircraft Co. and by Rockwell, and the mini-concentrator array design by Rockwell. Hughes is developing liquid phase epitaxy cells for 100-sun concentration, and Rockwell previously developed chemical vapor deposition cells for 400-sun concentration. Both types of arrays were designed for the mission orbits of 500 km and 1200 km for 35 kWe and 7 kWe power, respectively, and account for cell degradation with 7-year lifetime.


AMMONIA/ALUMINUM
DUAL SLOT HEAT PIPE
ALUMINUM FIN
.922
12.000
CONDENSER
CONDENSER
12.000
AMMONIA/ALUMINUM
DUAL SLOT HEAT PIPE
EVAPORATOR

ORC DUAL-SLOT RADIATOR PANEL CROSS-SECTIONS


12.000
.032
ALUMINUM FIN
METHANOL/TITANIUM
DUAL SLOT HEAT PIPE
CONDENSER
12.000
HEAT EXCHANGER
METHANOL/TITANIUM
DUAL SLOT HEAT PIPE
ALUMINUM SADDLE
EVAPORATOR

CBC METHANOL/TITANIUM RADIATOR PANEL CROSS-SECTIONS

Figure 6.2.1.6-1. Dual-Slot Heat Pipe Advanced Development Radiator Panels

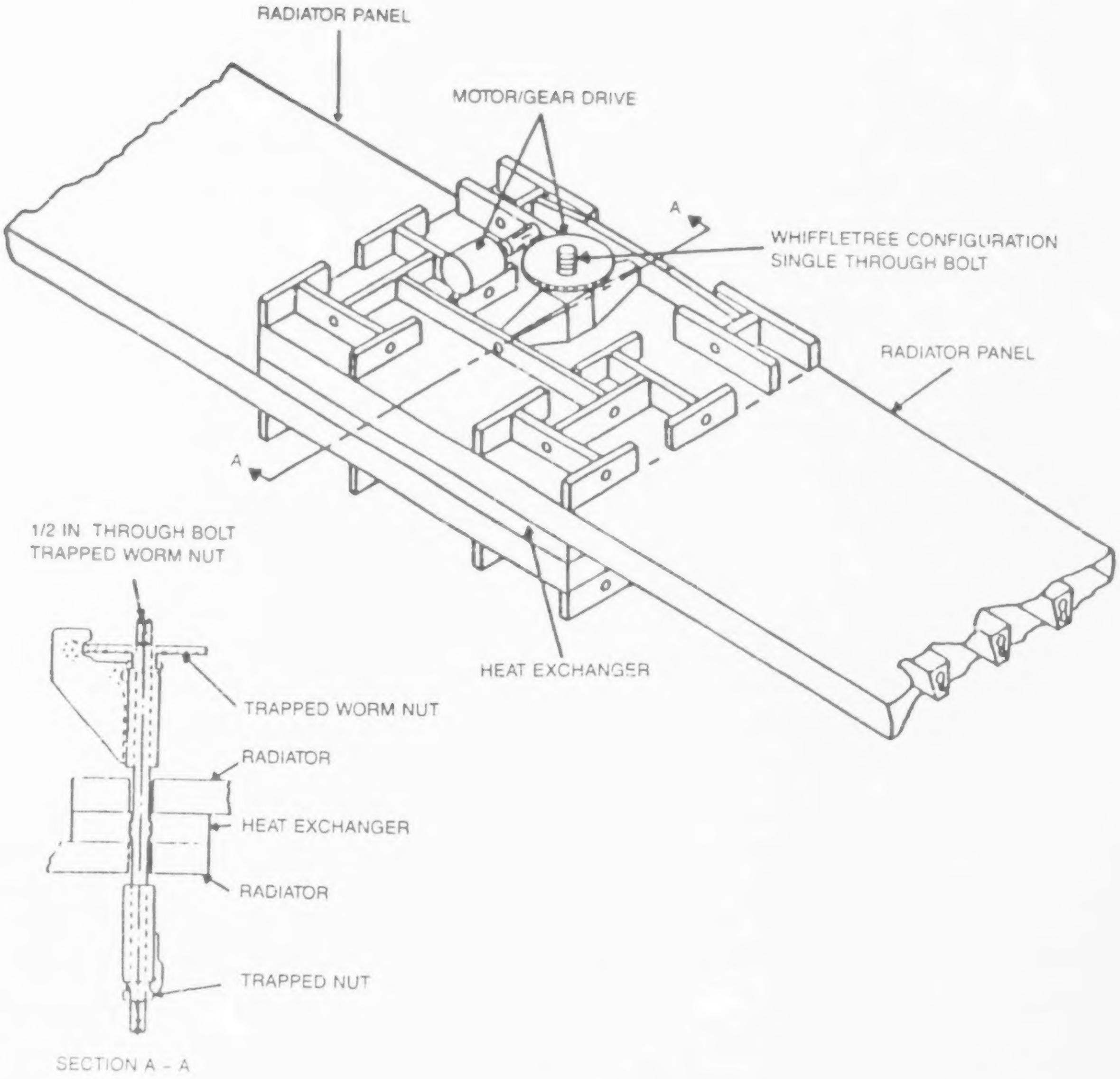

RADIATOR PANEL
MOTOR/GEAR DRIVE
A
WHIFFLETREE CONFIGURATION
SINGLE THROUGH BOLT
RADIATOR PANEL
A
1/2 IN. THROUGH BOLT
TRAPPED WORM NUT
HEAT EXCHANGER
TRAPPED WORM NUT
RADIATOR
HEAT EXCHANGER
RADIATOR
TRAPPED NUT
SECTION A - A

Figure 6.2.1.6-2. Whiffletree Advanced Development Radiator Clamping Device

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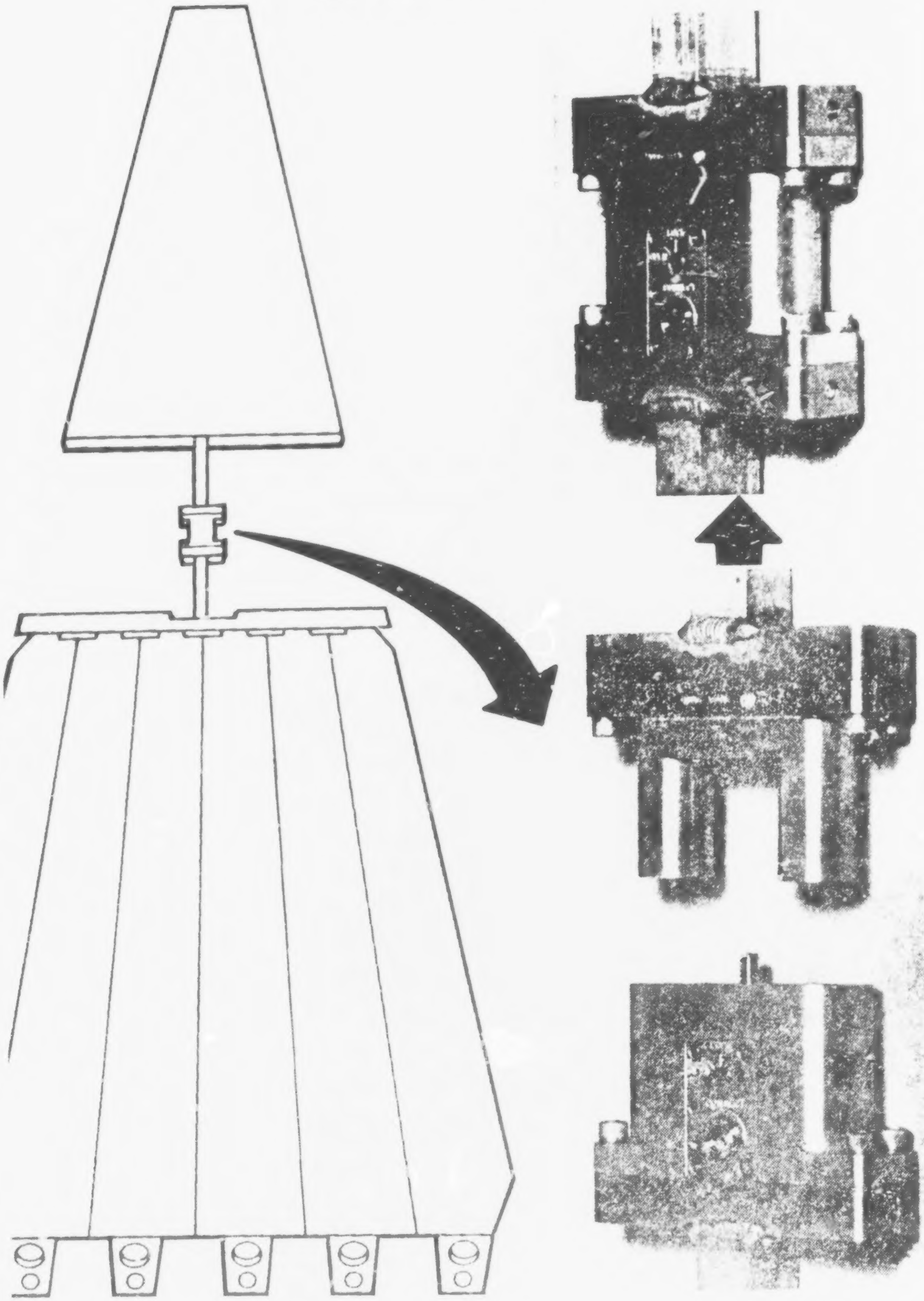
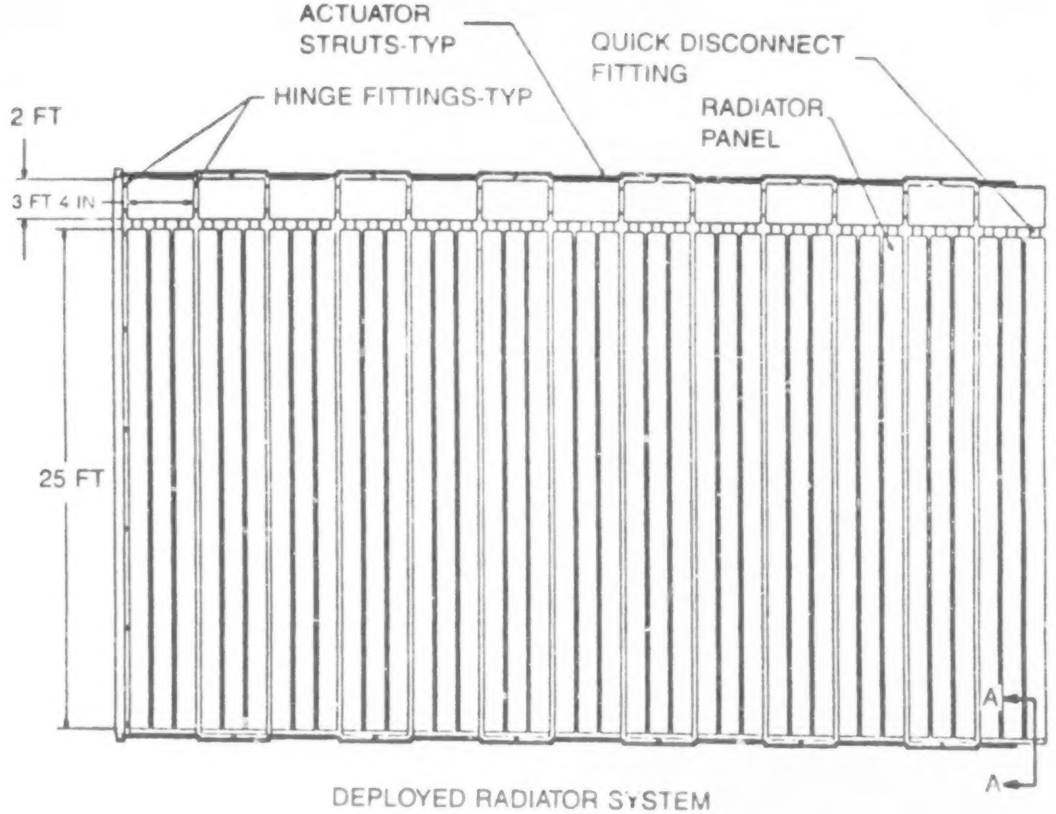
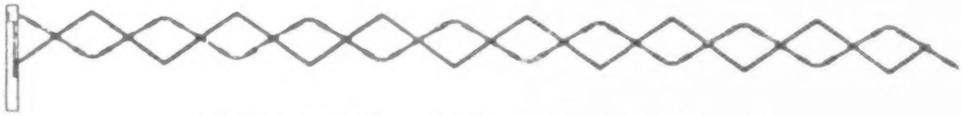


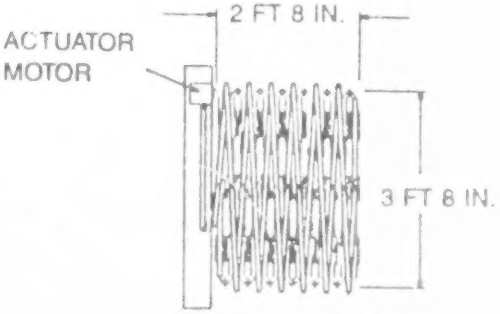
Figure 6.2.1.6-3. Heat Pipe Radiator Advanced Development Radiator Panels


ACTUATOR STRUTS-TYP
QUICK DISCONNECT FITTING
HINGE FITTINGS-TYP
2 FT
RADIATOR PANEL
3 FT 4 IN
25 FT
A
A

DEPLOYED RADIATOR SYSTEM



PARTIALLY DEPLOYED RADIATOR SYSTEM


2 FT 8 IN.
ACTUATOR MOTOR
3 FT 8 IN.

RADIATOR SYSTEM
IN STOWED POSITION

Figure 6.2.1.6-4. Deployable Heat Pipe Radiator

The silicon cell array is a more mature technology than that of the concentrator type array due to years of space experience, the recent shuttle test of the OAST-1 Experiment, the Solar Array Flight Experiment (SAFE) array configuration by Lockheed, and the present design work for Space Station. Near-term testing is planned for the GaAs concentrator with the Photovoltaic Array Space Power Experiment, which will have several different array designs, to be flown aboard the Shuttle by 1989 or later.

Trade results comparing the two types of arrays indicate the GaAs concentrator array to be about the same weight, but considerably smaller. Design maturity for the concentrator may tend to cause weight to increase with further design analysis; however, design innovation to devise a lighter weight collapsible structure could ultimately reduce weight.

### 6.2.3 Power Conversion State-of-the-Art

Rating of solar dynamic and photovoltaic power systems required selection of a common electrical output condition in order that the systems be compared on an equivalent basis. The Space Station Phase B trade studies considered 400 Hz and 20 kHz output, and the latter was selected. This will bring about a new standard, complementing the 28 V dc standard of past and present spacecraft experience and the 400 Hz standard for current military aircraft, thus ensuring development of 20 kHz equipment for future space use. Power conversion output conditions of 208 or 440 V ac, 20 kHz, single-phase power were chosen for this study.

The effect upon the power systems was to impose a power conversion efficiency and weight penalty upon each system. Output from the dynamic systems was put through an ac-ac converter for frequency conversion. In the case of the Stirling alternator output, any required power factor correction was assumed to be accomplished with a relatively small output capacitor or possibly by the ac-ac converter. The dc power output of the photovoltaic arrays was put through a dc-ac inverter. One advantage of the 20 kHz conversion equipment is that it will be lighter weight than for lower frequencies (i.e., 400 Hz) and will have slightly higher efficiency. The conversion efficiencies assumed for this study were 93% for ac-ac conversion and 91% for dc-ac inversion (see section 9).

Power output needs for future spacecraft will be varied, and in fact some power may be used in the as-produced condition. This is desirable where practical, as the power generating system must be oversized to account for conversion efficiency, and the waste heat generated in the conversion would normally require a sizable low-temperature radiator for heat rejection.

Handling equipment for 20 kHz power is presently undergoing development and test in support of the forthcoming Space Station program. Maturity is less than for the lower frequency equipment; however, the choice of 20 kHz does represent near term SOA.

## 6.3 ANALYTICAL TRADE STUDY METHODOLOGY

This section describes the program management approach, cycle computer codes, code validation, data and assumptions which went into the trade studies, and brief remarks on the objectivity of the study results.

### 6.3.1 Program Management

This program was performed jointly by two organizations within Rocketdyne, with program management provided by the Space Station Power Programs organization and the technical work carried out within the Advanced Programs organization. This matrix organization approach ensured that the project team worked closely with the Space Station team, and was cognizant of and utilized the rapidly evolving data base for Space Station photovoltaic and solar dynamic power systems. Performance of the program technical work within the Advanced Programs organization provided an independent perspective beyond the Space Station Power Programs work tasks and schedules, and avoided any possible conflict of interest between the two programs.

### 6.3.2 Description of Dynamic Cycle Computer Codes

The computer code descriptions presented herein are intended to indicate to what level of detail each of the codes describes a component or subsystem.

These remarks, along with those given in section 6.2.1 describing the technology base for the solar dynamic systems, are intended to adequately qualify the results of the trade studies presented in section 6.4.

### 6.3.2.1 Solar Concentrator Code

The offset parabolic truss hex concentrator configuration selected for the Space Station was represented by this code. Input to the code includes: f/D ratio for the offset parabola, reflectivity, surface error, pointing error, block/shade area, and the ratio of reflective area to hex planform area (packing factor). The equations used for cosine loss (relating surface area to aperture area) and for intercept factor were those for a symmetrical parabolic concentrator and do not accurately represent the offset concentrator. The cosine loss term was decreased to correspond to results of ray tracing analyses by GTRI (Georgia Tech Research Institute, ref. 3). The calculated intercept factor was used without any adjustment for the offset concentrator.

Weight algorithms of the form A * Area + B were used, which included the triangular reflective facets, hex panels, struts, latches, and controls. Area used was concentrator flat gross area, which is aperture plus block/shadow, increased by the cosine loss. The equation coefficients were:

| No. of Hexes | A $kg/m^2$ | B kg |
|---|---|---|
| 19 | 3.265 | 131.5 |
| 7 | 3.153 | 64.6 |

The splined radial panel (SRP) concentrator was also examined; however, this concentrator is of a symmetrical parabolic configuration (ref. 3). A similar equation was used for the SRP weight, although area was maximum aperture area (aperture plus block/shadow).

Weight = 1.41 * Area + 46 kg

The SRP weight includes the complete concentrator with deployment mechanism, struts, controls, and 20 kg contingency. The coefficients are appropriate for a 35 kWe power system, and could be inaccurate at 7 kWe. The effect of a lighter weight concentrator such as the SRP is to permit increased concentrator area and reduced radiator area (thereby increasing radiator temperature) for the case of minimum power system weight for either the CBC or Stirling cycles. Any increase in radiator temperature would require use of water heat pipes rather than methanol, as the baseline truss hex power system designs were found to already be limited by methanol temperature.

### 6.3.2.2 Receiver/Thermal Energy Storage Code

The code was developed to represent receiver/TES designs in a generic fashion, with sizing accomplished by scaling existing designs rather than designing from fundamentals. The code was used for both receiver designs with TES internal to the receiver cavity (such as the CBC or Rankine designs) and for the pumped loop and heat pipe receivers wherein the TES was located externally. The code logic for the sizing process is as follows:

1. Determine thermal energy storage required for eclipse based on engine requirements, estimated receiver and TES losses, and an oversizing margin.
2. Heat absorbing tubes of specified diameter and spacing (whether straight or coiled tubing) are assumed to form a right cylinder of specified L/D. Sizing of the right cylinder is dependent on either specified heat flux allowable on the tubes, or on the number of tubes needed to contain the TES material. Input values of individual overall tube volume, and of tube weight, each as a ratio of TES material volume, are used in the sizing calculations.
3. The receiver shell is sized based on specified L/D, clearance from the tube bundle, wall thickness, wall density, and aperture shield density.
4. Energy losses by conduction (including TES), reradiation, and reflection are updated and the receiver sizing process is reiterated from step 1 several times to obtain convergence.

5. The total energy requirements are then used as input to size the concentrator.

### 6.3.2.3 Power Conversion Unit Codes

The CBC PCU was patterned closely after the Space Station CBC design (ref. 1). The code is quite detailed in the sizing of hardware, including detailed design of the turbine and compressor and requires a large number of user inputs. Included in these are: power, speed, working fluid composition, pressure, pressure ratio, pressure drops allowable, maximum and minimum cycle temperatures, and detailed parameters for each component. Outputs are PCU subsystem and component weights, sizes, efficiencies, and performance. User options include:

1. One of three alternator types
2. Axial or radial turbine
3. Axial or centrifugal compressor
4. One of two recuperator heat exchanger designs
5. One of two waste heat exchanger designs
6. Specification of various bypass flows (turbine and/or recuperator) for engine control, and lesser flows for cooling the alternator and shaft bearings

The Stirling PCU was represented by the following equations for the free piston Stirling engine with linear alternator:

1. For system efficiency (as fraction of Carnot) where system efficiency is electrical power out : heat input:

   FCEFF = (1 + FXE) * 0.6 * (TRATIO - 1.1575) / (TRATIO - 1.00)

   FXE: fractional improvement in efficiency

   TRATIO: temperature ratio, $T_H/T_C$
2. For specific mass (kg/kWe):

   SPECM = (1 - FXSM) * (5.96 - (0.8756 / (TRATIO - 1.1575)) +
   (0.7732 / (TRATIO - 1.1575)$^2$))

   FXSM: fractional improvement in specific weight

The following coefficients were obtained from the data presented in reference 5.

| Nominal Power, kWe at TRATIO = 2.0 | At TRATIO = 2.0 FCEFF | At TRATIO = 2.0 SPECM | FXE | FXSM |
|---|---|---|---|---|
| 8 | 0.572 | 7.0 | 0.132 | 0.165 |
| 25 | 0.587 | 5.5 | 0.161 | 0.085 |
| 40 | 0.587* | 5.5 | 0.161 | 0.085 |

*Note: Efficiency of 40 kWe set equal to the value at 25 kWe.

The foregoing expressions were initially used for Stirling engine characterization at both the 7 kWe and 35 kWe sizes; however, more extensive data was later obtained from MTI for the 25 kWe size SSE engine, which provided an interrelationship between engine efficiency and weight for a range of temperature ratios. This information, shown previously in figure 6.2.1.4-2, was finally used to represent the 35 kWe engine for this study.

### 6.3.2.4 Radiator Code

The radiator code was designed to represent the heat pipe type of radiator in a generic fashion by input of characteristics of existing radiator designs. Both the CBC and the Stirling PCU designs utilize pumped liquid loops for waste heat removal from the engine (3M FC-75 fluorinated organic liquid for the CBC and NaK78 liquid metal eutectic for the Stirling). The coolant is pumped through the engine and then to the heat exchanger boom of the radiator where the waste heat is transferred to the heat pipe evaporator sections.

The heat pipe condenser code analyzes a thermal symmetric condenser section by dividing the fin into ten longitudinal strips so as to calculate the fin temperature profile from root to tip. This is repeated for a number of heat pipes until the rejected heat is equal to or greater than the amount of waste heat which needs to be rejected. This whole number of heat pipes is then reduced by a fraction for purposes of radiator area and weight calculations, so as to avoid step-function changes in radiator weight during the course of power system trade studies.

The data input to the code was derived from reference 2. Inputs include emissivity, number of views per fin, sink temperature, thermal resistances between the coolant and the evaporator, specific weights of the boom heat exchanger including heat pipe attachment, evaporator, condenser, and coolant return line. The code is capable of analyzing a hybrid radiator with two different types of heat pipes (e.g. titanium/methanol and aluminum/ammonia).

### 6.3.3 Code Validation

The code validation process was carried out early in the study. The subsystem codes were run for Space Station CBC operating conditions and adjustments made in the code and input coefficients so as to obtain good correlation. The initial validation was conducted using the 12/85 edition of the Space Station DR-02 document (see reference 1), and was continued as the later editions of DR-02 became available. As was mentioned in section 6.3.2, the receiver/TES, and radiator codes were written to require input from existing component designs so as to use scaling methods for alternative sizing conditions.

### 6.3.4 Data and Assumptions

This section summarizes the basic data sources and the assumptions that went into the conduct of the analytical characterization and comparison (trade study) of the solar dynamic power cycles. Much of the data was obtained from the Space Station Phase B analyses and trade studies. Revisions to the data as may have occurred during the Space Station Phase C/D proposal preparation were not made available for incorporation into this study.

### 6.3.4.1 Solar Concentrator

The baseline concentrator configuration was chosen as the truss hex concentrator as proposed for Space Station. Data was obtained from the 12/86 DR-02 (ref. 1). An alternate concentrator configuration was also examined, the splined radial panel (see section 6.4.4).

C-2

The 2-axis vernier pointing gimbal (532 kg, ref. 1) was deleted from the concentrator, as the method of fine pointing would probably be different for different spacecraft. Block/shadow area was reduced from reference 1 (13 $m^2$ for Stirling and 15 $m^2$ for CBC, at 35 kWe power) due to elimination of the vernier pointing gimbal, thereby avoiding the misalignment potential of the radiator shadow upon the concentrator. Packing factor for the truss hex was held constant at 0.935. Reflectivity was chosen as 0.90 for 500 km orbital altitude (due to atomic oxygen degradation) and 0.93 for 1200 km orbital altitude at 7-year lifetime. The nominal pointing error was chosen as 0.1° and surface accuracy was chosen as 2.5 mrad. The truss hex concentrator cosine loss term was reduced from the code calculated value of 0.1795 to the GTRI ray tracing derived value of 0.1012 (ref. 3), each at a concentrator f/D of 0.25. Latches and mounting hardware were included in the concentrator fixed weight (60 required for the 19-hex arrangement and 18 required for the 7-hex arrangement). Concentrator fixed weights also include the concentrator control computer and two sun sensors but exclude the motor controllers and wiring harness.

The various correction terms for the splined radial panel (SRP) concentrator: reflectivity, packing factor, and intercept factor were judged to collectively be about equivalent to the truss hex design. Block/shadow area for the SRP was judged to be about the same as for the truss hex, as the SRP receiver/TES and PCU blockage would be on the order of the facet blockage loss (~3%) for the truss hex. Strut weight was reduced in proportion to the concentrator weight. The same weight was used for controls and sensors, and a 20 kg contingency weight was added. The SRP concentrator design (ref. 3) includes packaging and deployment mechanism in the concentrator weight. In the case of the SRP, being a symmetric concentrator, the weight algorithm was based on maximum aperture area rather than gross flat area.

#### 6.3.4.2 Brayton Power Conversion Unit and Solar Receiver

The solar receiver/TES configuration was a hybrid design employing the internal configuration designed by Boeing (L.M. Sedgwick, Boeing, Contract NAS3-24669) consisting of 24 4-inch diameter TES tubes, 60 inches long with 4-inch spacing. The 4-inch corrugated TES containment tube is attached to a

2-inch diameter heat exchanger tube, as shown in figure 6.2.1.2-2. The receiver shell configuration was similar to the Garrett Space Station CBC design. Data for the Boeing receiver intervals were as follows:

| | |
|---|---|
| Salt volume, $m^3$ | 0.1585 |
| Volume fraction nickel felt (using 20% dense felt) | 0.1875 |
| Containment volume, $m^3$ | 0.1950 |
| Felt metal weight, kg | 313 |
| Containment weight, kg | 187 |
| HX/manifold tubing, kg (estimated for flight weight) | 154 |
| Total TES weight without salt, kg (including HX/manifold tubing) | 654 |
| Ratio: total TES weight without salt to salt volume, $kg/m^3$ | 4130 |

The foregoing data formed the basis for the CBC receiver/TES sizing, with adjustments for TES material quantity and the smaller 7 kWe receiver size. Data for the receiver/TES were as follows:

| | 35 kWe | 7 kWe |
|---|---|---|
| Nickel felt density* | 20% | 20% |
| Ratio: total TES weight without salt to salt volume, $kg/m^3$ | 4130 | 3960 |
| TES outer tube diameter, cm (in.) | 10.16 (4.0) | 8.89 (3.5) |
| HX tube diameter, cm (in.) | 5.08 (2.0) | 3.81 (1.5) |
| Tube spacing, cm (in.) | 10.16 (4.0) | 8.89 (3.5) |
| Tube circle diameter, cm (in.) | 137.3 (54.0) | 66.8 (26.3) |
| Tube circle L/D | 1.04 | 1.04 |
| Receiver shell L/D | 1.14 | 1.14 |
| Reradiation temperature factor | 1.033 | 1.033 |

*Note. Felt metal does not extend into corrugations of TES outer tube.

Excess TES margin was 5%, and TES fill temperature was 1230K (1750F). The receiver shell was adjusted for the LiF temperature (1121K) by increasing the multi-foil insulation (MFI) from 0.3 inch to 0.4 inch thickness. MFI for the Space Station CBC design considered 0.0127 mm (0.0005 in.) foil with ten outer layers of aluminum foil and the remainder of nickel. For this study, the nickel was replaced with molybdenum foil due to higher receiver temperatures. MFI thickness was increased to 0.55 inch and 0.65 inch respectively for NaF and $Mg_2Si$ TES materials. The 0.5 inch inner formed insulation and the 0.1 inch aluminum outer shell were unchanged. Density for this composite construction with 0.4 inch MFI was 741 $kg/m^3$, thermal conductivity was 0.0100 W/mK. Receiver outer surface emissivity was assumed as 0.35 and orbit average sink temperature was assumed as 255K for the receiver surface radiation calculation. The reradiation temperature factor was the ratio of the orbital average cavity temperature to LiF melting temperature. Receiver reradiation calculations assumed an effective emissivity of 1.0.

Reflective losses from the receiver cavity were calculated as:

$$Q_L = Q_R * C * (1-\epsilon) \frac{(\pi/4) \quad (D_A^2)}{(\pi/4)\ (D_C^2 + 4\ D_C\ L_C)}$$

Where:

| | |
|---|---|
| $Q_L$ | reflective loss |
| $Q_R$ | energy entering receiver aperture |
| $C$ | approximately 1.8-2.0, estimated from Boeing data |
| $\epsilon$ | receiver inner surface reflectivity of solar spectrum<br>(~0.7 for metals, ~0.3 for non-metallics) |
| $D_A$ | aperture diameter |
| $D_C$ | cavity inner diameter |
| $L_C$ | cavity inner length |

The value of $C * (1-\epsilon) = 0.6$ was used for all receiver configurations, CBC and Stirling.

The aperture plate and shield combined specific weight was 45 kg/m$^2$ for a 0.050 inch aperture plate and 1.0 inch graphite shield. The shield thickness was a compromise between the Garrett design at 0.5 inch and the Boeing design at 2 inches. The shield diameter was chosen to be 15% larger than the receiver shell outside diameter.

The principal assumptions for the CBC PCU were:

<table>
<tr><th></th><th>35 kWe</th><th>7 kWe</th></tr>
<tr><td>Working fluid molecular weight</td><td>39.9</td><td>48.4</td></tr>
<tr><td>Speed, rpm</td><td>32,000</td><td>48,000</td></tr>
<tr><td>Alternator efficiency</td><td colspan="2">0.934</td></tr>
<tr><td>Turbine inlet temperature, K</td><td colspan="2">1086</td></tr>
<tr><td>Recuperator effectiveness</td><td colspan="2">0.94</td></tr>
<tr><td>Recuperator ΔP, fraction of inlet P</td><td colspan="2"></td></tr>
<tr><td>cold side</td><td colspan="2">0.0074</td></tr>
<tr><td>hot side</td><td colspan="2">0.0134</td></tr>
<tr><td>Compressor inlet pressure, kPa</td><td colspan="2">160</td></tr>
<tr><td>Compressor pressure ratio (PR)</td><td colspan="2">1.88</td></tr>
<tr><td>Pressure ratio factor (turbine PR ÷ compressor PR)</td><td colspan="2">0.934</td></tr>
</table>

### 6.3.4.3 Heat Pipe Stirling Power Conversion Unit and Receiver

The solar receiver/TES configurations shown in figures 2.4-2 and 6.2.1.4-6 employ the heat pipe/TES modules illustrated in figure 6.2.1.4-7. Data for the receiver/TES designs were as follows:

| | 35 kWe | 7 kWe |
|---|---|---|
| Nickel felt density | 16% | 16% |
| Ratio: module weight* without salt to salt volume, kg/m3 | 5540 | 5540 |
| Ratio: including insulation, kg/m3 | 6200 | 6690 |
| Heat pipe tube diameter, cm (in.) | 3.81 (1.50) | 3.81 (1.50) |
| Tube spacing, cm (in.) | 1.78 (0.70) | 1.27 (0.50) |
| Tube circle diameter, cm (in.) | 71.1 (28.00) | 33.6 (13.25) |
| Tube circle L/D | 1.0 | 1.0 |
| Receiver shell L/D | 1.1 | 1.1 |
| Reradiation temperature factor | 1.021 | 1.021 |
| Insulation area ratio | 2.5 | 2.9 |

*Note: Module weight includes weight of both primary and secondary heat pipes (except secondary condenser) plus felt metal.

Excess TES margin was 2.5%, and TES fill temperature was 1230K (1750F).

The same receiver shell construction was used for the Stirling design as was used for CBC (section 6.3.4.2). This same construction was also assumed for insulation of the TES section and the hot end of the engine, which increased total conduction loss area as indicated by the insulation area ratio (above). The 1.0 inch graphite shield was used for the Stirling design, with a diameter equal to the insulated TES section outer diameter.

The principal assumptions for the heat pipe Stirling PCU were:

| | 35 kWe | 7 kWe |
|---|---|---|
| Nominal (design) hot temperature ($T_H$), K | 1033 | 1033 |
| Maximum allowable hot temperature, K | 1050 | 1050 |
| Nominal temperature ratio ($T_H/T_C$) | 2.7 | 2.9 |
| Engine specific weight, kg/kWe (gross) | 8.0 | 6.7 |
| Weight multiplier for excess power | 1.067 | 1.18 |
| Adjusted specific weight, kg/kWe (gross) | 8.5 | 7.9 |

Maximum allowable hot temperature was based on material properties limitations for the SSF design, and nominal temperature was determined by the thermal transient analysis (section 12). Temperature ratios were chosen so as to not exceed methanol heat pipe maximum temperature for maximum solar input conditions. The engine specific weight for 35 kWe results in near-minimum power system weight. The specific weight for the 7 kWe engine corresponds to reference 5 results. The weight multiplier for excess power increases alternator weight (assumed as 1/3 of engine weight) in proportion to the excess power which would occur at maximum solar input.

#### 6.3.4.4 Pumped Loop Stirling Power Conversion Unit and Receiver

The solar receiver/TES configuration, as shown in figure 6.2.1.4-5 for the 35 kWe power level, utilizes a liquid metal pumped loop for heat transport. Heat is absorbed by one or more helically-wound coiled tubes which form a right cylinder plus a few turns at the back of the receiver. The smaller receiver coil was formed from a single tube wound into a coil, whereas the larger receiver coil was wound using two tubes beginning 180° apart (forming a helix arrangement similar to a double threaded screw). The sizing of the coil was a balance of several considerations:

1. Coolant velocity <3 m/sec (10 ft/sec)
2. Minimize tube diameter to reduced coolant inventory
3. Average heat flux <200 $kW/m^2$ (projected area = $D_{tube} * L_{tube}$)

The remote TES is similar to a shell and tube heat exchanger, with the tubes being sealed cannisters of TES material. The concern of void formation and control was not definitively resolved; therefore, the weight of the TES containment remains correspondingly uncertain. To provide allowance for void management, whether by the approaches discussed in section 10.4.3 or by other approaches, the weight (wall thickness) of the originally designed 1 inch diameter by 0.050 inch wall cannisters was doubled. The weight of the TES assembly was evaluated with 295K (70F) density NaK, as were other NaK containing components, rather than at operating temperatures; therefore, the hot and cold accumulator weights were indicated as dry weights.

Data for the receiver and TES design were as follows:

|  | 35 kWe | 7 kWe |
|---|---|---|
| Ratio: TES weight with NaK @ 70F and without salt to salt volume, $kg/m^3$ | 6830 | 7720 |
| Ratio: including insulation, $kg/m^3$ | 7030 | 8120 |
| Tube diameter, cm (in.) | 3.18 (1.25) | 2.54 (1.00) |
| Tube spacing, cm (in.) | 1.27 (0.50) | 1.27 (0.50) |
| Number of tubes in helix coil | 2 | 1 |
| Approximate number of backwall turns | 2 | 0 |
| Tube circle diameter, cm (in.) | 61.4 (24.20) | 33.5 (13.20) |
| Tube circle L/D | 1 | 1 |
| Receiver shell L/D | 1.1 | 1.1 |
| Reradiation temperature factor | 0.991 | 0.991 |
| Insulation area ratio | 2.8 | 3.4 |

Excess TES margin was 4%, and TES fill temperature was 1230K (1750F).

The same receiver shell construction was used for the Stirling design as was used for CBC (section 6.3.4.2). This same construction was also assumed for insulation of the TES vessel and the hot end of the engine, which increased total conduction loss area as indicated by the insulation area ratio (above). The 1.0 inch graphite shield was used for the Stirling design, in a rectangular shape so as to shield the receiver, TES, and other components. The principal assumptions for the pumped loop Stirling PCU were:

|  | 35 kWe | 7 kWe |
|---|---|---|
| Nominal (design) hot temperature ($T_H$), K | 1045 | 1045 |
| Maximum allowable hot temperature, K | 1050 | 1050 |
| Nominal temperature ratio ($T_H/T_C$) | 2.73 | 2.93 |
| Engine specific weight, kg/kWe (gross) | 8.0 | 6.7 |
| Weight multiplier for excess power | 1.067 | 1.18 |
| Adjusted specific weight, kg/kWe (gross) | 8.5 | 7.9 |

Temperature ratios were chosen to result in the same values of $T_C$ as for the heat pipe Stirling designs.

### 6.3.4.5 Waste Heat Radiator

The Grumman dual-slot heat pipe radiator (ref. 2) was chosen as the baseline radiator. Two types of heat pipe panels were considered, as follows:

|  | Aluminum | Titanium |
|---|---|---|
| Material: pipe | Aluminum | Titanium |
| fin | Aluminum | Aluminum |
| Working fluid | Ammonia | Methanol |
| Width, m (ft) | 0.305 (1.00) | 0.305 (1.00) |
| Evaporator length, m (ft) | 0.610 (2.00) | 0.610 (2.00) |
| Number evaporator legs | 8 | 10 |
| Condenser length, m (ft) | 13.72 (45.0) | 7.62 (25.0) |
| Number condenser legs | 2 | 2 |
| Pipe inside diameter, cm (in.) | 1.90 (0.75) | 1.90 (0.75) |
| Pipe wall thickness, cm (in.) | 0.254 (0.100) | 0.152 (0.060) |
| Fin construction | Monocoque | Wing |
| Total fin thickness, cm (in.) | 0.081 (2*0.016) | 0.081 (0.032) |
| Panel weight, kg (lb) | 35.7 (78.8) | 18.6 (41.1) |

The different types of fin construction are shown in figure 6.2.1.6-1; the monocoque being the box-like shape shown for the aluminum/ammonia panel, and the wing being the single sheet fin bonded to the condenser tubes shown for the titanium/methanol panel.

Several variations in the heat exchanger boom design were considered: both single sided and double sided configurations, constructed of aluminum or titanium, with FC-75 or NaK coolant. The single sided configuration has panels extending only from one side of the boom, whereas the double sided configuration has panels extending from both sides of the boom (see fig.

2.4-1). Various attachment options were also considered. Boom weights included heat exchanger, coolant fluid, attachment device, and an assumed 7.4 kg/m (5 lb/ft) weight allowance for structural strongback. Coolant fluid weights were calculated for 294K (70F). The following combinations of the above options were selected:

| | CBC | Stirling |
|---|---|---|
| Coolant | FC-75 | NaK |
| Specific gravity at 294K | 1.730 | 0.868 |
| Heat exchanger material | Aluminum | Titanium |
| | 35 kWe | 7 kWe |
| Heat exchanger configuration | Double sided | Single sided |
| Attachment option | Whiffletree | Quick disconnect |
| Number of panels attached | 2 | 1 |
| Attachment weight, kg (lb) | 13.9 (30.7) | 6.2 (13.6) |

The resulting heat exchanger boom weights used in the study were:

| | CBC | | Stirling | |
|---|---|---|---|---|
| | 35 kWe | 7 kWe | 35 kWe | 7 kWe |
| Heat exchanger (wet), lb/ft | 21.4 | 12.3 | 24.3 | 14.8 |
| Attachment, lb/ft | 30.7 | 13.6 | 30.7 | 13.6 |
| Strongback (assumed), lb/ft | 5.0 | 5.0 | 5.0 | 5.0 |
| Total weight, lb/ft | 57.1 | 30.9 | 60.0 | 33.4 |
| Total weight, kg/m | 85.0 | 46.0 | 89.3 | 49.7 |

The coolant return line and fluid weights were calculated with stainless steel tubing. The sizes and weights were:

| | CBC | | Stirling | |
|---|---|---|---|---|
| | 35 kWe | 7 kWe | 35 kWe | 7 kWe |
| Tubing outside diameter, in. | 0.75 | 0.50 | 2.00 | 1.25 |
| Tubing wall thickness, in. | 0.083 | 0.065 | 0.083 | 0.083 |
| Line (tubing and shield) weight*, lb/ft | 1.16 | 0.45 | 3.14 | 1.62 |
| Line weight*, kg/m | 1.73 | 0.67 | 4.67 | 2.40 |

*Note: Includes 15% margin for fittings.

### 6.3.4.6 Power Conversion to 20 kHz

Solar dynamic power was converted to 20 kHz output at a 93% efficiency. Weight allowance was made for the converter, controllers (2), and for the electronic component cooling (ECC) auxiliary radiator. Thermal load on the ECC radiator was primarily from the frequency converter, but included other electronic cooling loads as well:

| | 35 kWe | 7 kWe |
|---|---|---|
| Frequency conversion, kW | 3.425 | 0.705 |
| Engine controller PLR, kW | 0.375 | 0.125 |
| Engine control computers (2) and pointing control, kW | 0.450 | 0.360 |
| Total, kW | 4.25 | 1.19 |

The titanium/methanol heat pipe panel was chosen to cool a cold plate with an assumed 294K (70F) cold plate interface temperature. (Note: The Space Station Integrated Thermal Control (ref. 1) was designed for a 278K (5C) cold plate interface termperature for PV system battery cooling, whereas an interface temperature of 293K (20C, or 68F) was adequate for PMAD components.) The ECC radiator sizing was as follows:

| | 35 kWe | 7 kWe |
|---|---|---|
| Heat to be rejected, kW | 4.25 | 1.19 |
| Estimated heat rejection per panel, kW | 1.045 | 1.085 |
| Number of panels required | 4.07 | 1.10 |
| Number of panels including redundancy | 5 | 1.37 |
| ECC radiator weight (panels, boom, line, fluid, 10% margin), kg | 178 | 50 |

### 6.3.4.7 Parasitic Power

Parasitic power requirements were derived from the CBC design for Space Station (ref. 1). The largest element relates to the minimum nominal power required for the parasitic load radiator (PLR) and engine controllers for power management and control. The actual parasitic power elements are drawn from different locations and conditions in the electrical circuit; however, the total is expressed as if drawn from the 20 kHz converter net output so as to satisfy input conditions to the code. The following parasitic loads were used:

|  | 35 kWe | 7 kWe |
|---|---|---|
| Engine control (switching electronics), W | 375 | 100 |
| Minimum nominal PLR load, W | 600 | 150 |
| Engine controllers (2), W | 220* | 150* |
| Pointing controllers (2) and motor power, W | 280* | 200* |
| Total parasitic power, W (gross) | 1475 | 600 |
| Power conversion efficiency | 0.93 | 0.93 |
| Total parasitic power, W (net) | 1370 | 560 |

*Note: Gross power based on dedicated converter with 0.9 efficiency.

The Stirling power cycles utilize low-efficiency EM pumps in pumped heat transport loops whereas the CBC cycle utilizes a canned-rotor centrifugal pump. Efficiencies of the EM pumps are somewhat uncertain, as was discussed in section 6.2.1.5, especially for the lower flowrate designs. Furthermore, it is not known whether the EM pumps will require dedicated power converters or not. The following net power requirements were used to provide for the heat transport loop pumping needs:

|  | 35 kWe | 7 kWe |
|---|---|---|
| CBC, W | 240 | 75 |
| Heat pipe Stirling, W | 650 | 470 |
| Pumped loop Stirling, W | 1190 | 775 |

### 6.3.5 Study Objectivity

Evidence of study objectivity lies in source and interpretation of the assumptions which went into the study. Common computer codes were used for all but the PCU characterization. The same configuration was used for both concentrator and radiator. The CBC PCU was validated by comparison to the proposed Space Station CBC PCU, a fairly easy task as most of the data characterizing the PCU components were obtained from the Space Station design. The Stirling PCU performance was obtained from NASA-LeRC subcontractors, and was reviewed and approved by NASA for application to this study. The CBC cycle technology is more mature than the free piston Stirling engine. As a result, more technology tasks have been identified and recommended so as to close that technology gap. Performance of these and other technology investigations will in time provide confirmation of whether the power system performance gains predicted herein are to be realized.

## 6.4 ANALYTICAL TRADE STUDY RESULTS

This section presents the results of the analytical trade study. Tables of weights, areas, and selected other design details comparing the six solar dynamic power system designs are presented. In addition, the solar dynamic and photovoltaic power systems are compared.

Many design detail trades were performed in support of the design point selection process for the different power system designs. A series of such results are presented for both the 35 kWe CBC and heat pipe Stirling power systems, which serve to illustrate the effect of various parameters upon the system (primarily expressed as weight and/or area). The general objective was to minimize system weight; however, since this was a conceptual design study without specific application, further optimization reiterations were not performed.

### 6.4.1 Comparison of Power System Design Results

This section presents tabular comparisons of the solar power system designs. Table 6.4.1-1 presents a summary comparison of selected parameters for the CBC design and heat pipe Stirling design (the better of the two Stirling designs) for both the 35 kWe and 7 kWe power missions. Overall power system size is presented in the figures of section 7. Table 6.4.1-2 presents a summary comparison of the silicon and gallium arsenide photovoltaic designs for both the 35 kWe and 7 kWe power missions. Additional information on the photovoltaic designs is presented in section 9.

Table 6.4.1-3 presents power system weight breakdowns for each of the six solar dynamic power systems. Table 6.4.1-4 presents a variety of other power system parameters, including area, efficiency, parasitic power, etc.

State point diagrams for each of the six solar dynamic power system designs are presented in figures 6.4.1-1 through 6.4.1-6. Each diagram presents a schematic and a listing of various parameters such as power, temperature, pressure, and flowrate about the power system.

Table 6.4.1 1. Solar Dynamic Power Systems Design Data Summary

| Parameter | 35 kWe | | 7 kWe | |
|---|---|---|---|---|
| | Brayton | Stirling(1) | Brayton | Stirling(1) |
| Concentrator gross aperture area, $m^2$ (2) | 196 | 168 | 37.4 | 30.4 |
| Radiator radiant area, $m^2$ (3) | 211 | 137 | 58.8 | 40.8 |
| Radiator sail area, $m^2$ | 110 | 71 | 31.7 | 22.2 |
| Solar multiple (4) | 1.607 | 1.607 | 1.467 | 1.467 |
| Excess energy ratio(5) | 1.22 | 1.22 | 1.57 | 1.57 |
| PCU plus alternator efficiency | 0.356 | 0.420 | 0.342 | 0.408 |
| Efficiency - solar to net power | 0.217 | 0.253 | 0.207 | .235 |
| Concentrator, kg | 845 | 742 | 196 | 180 |
| Receiver/TES, kg | 1255 | 1075 | 281 | 254 |
| PCU, alternator, control (PLR), and structure, kg | 878 | 574 | 266 | 158 |
| PCU radiator and electronic cooling radiator, kg | 1471 | 1006 | 421 | 309 |
| Pumps, accumulators, piping and fluid alowance, kg(6) | 78 | 74 | 18 | 40 |
| Power conversion to 20 kHz, kg | 200 | 200 | 134 | 134 |
| Interface structure, kg(7) | 340 | 268 | 102 | 83 |
| Power system weight, kg | 5067 | 3939 | 1418 | 1158 |

Notes

1. Heat pipe Stirling configuration
2. Includes blockage and shadow area, and hex segment packing factor (reflective facet area ÷ hex area)
3. Brayton cycle PCU waste heat and electronic cooling loads are combined and serviced by a single radiator. Stirling cycle PCU waste heat load and electronic cooling load are serviced by separate radiators due to temperature differences. Areas include approximately 15% redundancy for seven year lifetime.
4. Orbit period ÷ shortest sun interval for the orbit
5. Orbit (maximum solar intensity times longest sun interval) ÷ (minimum solar intensity times shortest sun interval)
6. Primarily required for the PCU waste heat transport loop. The Stirling cycle utilizes NaK liquid metal for heat transport, and requires an electromagnetic pump.
7. Mounting structure for attachment of the various subsystems including beta-joint interface ring

Table 6.4.1-2. Photovoltaic Power System Summary

| Parameter | 35 kWe | | 7 kWe | |
|---|---|---|---|---|
| | Silicon Planar | GaAs Concentrator | Silicon Planar | GaAs Concentrator |
| Design EOL power, kWe | 35.0 | 35.0 | 7.00 | 7.00 |
| Array dc power output, kWe[1] | 74.4 | 74.4 | 13.98 | 13.98 |
| Continuous dc power output, kWe | 38.5 | 38.5 | 7.69 | 7.69 |
| Frequency inverter efficiency, dc/ac | 0.91 | 0.91 | 0.91 | 0.91 |
| Array active panel area, $m^2$ | 741 | 366 | 142 | 72 |
| Array sail area, $m^2$ [2] | 933 | 407 | 182 | 80 |
| Array assembly weight, kg[1,2] | 1578 | 1505 | 317 | 297 |
| Energy storage subsystem weight, kg | 2373 | 2373 | 571 | 571 |
| Thermal control subsystem weight, kg | 1633 | 1633 | 464 | 464 |
| Power management and distribution weight, kg | 1339 | 1339 | 448 | 448 |
| Total PV power system weight, kg | 6923 | 6850 | 1800 | 1780 |

Notes:
1. Includes sequential shunt unit (SSU)
2. Includes dummy panels, array containment box, mast, etc.

Table 6.4.1-3. Solar Dynamic Power Systems Weight Comparison

| Parameter | 35 kWe | | | 7 kWe | | |
|---|---|---|---|---|---|---|
| | Brayton | Heat Pipe Stirling | Pumped Loop Stirling | Brayton | Heat Pipe Stirling | Pumped Loop Stirling |
| Power system weight, kg | 5067 | 3939 | 4485 | 1418 | 1158 | 1341 |
| Concentrator, kg | 845 | 742 | 743 | 196 | 180 | 183 |
| Receiver/TES, kg | 1255 | 1075 | 1308 | 281 | 254 | 326 |
| Phase change material, kg[1] | 266 | 219 | 221 | 56.0 | 47.5 | 49.3 |
| PCU with alternator, kg | 572 | 339 | 344 | 163 | 68 | 70 |
| PCU mounting structure, kg | 146 | 75 | 200 | 33 | 20 | 55 |
| Electric loop control (PLR), kg | 160 | 160 | 160 | 70 | 70 | 70 |
| PCU radiator, kg[2] | 1471 | 828 | 827 | 421 | 259 | 269 |
| EM pumps, kg | 0 | 28 | 72 | 0 | 20 | 42 |
| Accumulators plus piping and fluid allowance, kg | 78 | 45 | 150 | 18 | 20 | 46 |
| Frequency converter and controllers, kg | 200 | 200 | 200 | 134 | 134 | 134 |
| Electronic components cooling radiator, kg | (2) | 178 | 178 | (2) | 50 | 50 |
| Interface adaptor and superstructure, kg | 340 | 268 | 303 | 102 | 83 | 96 |

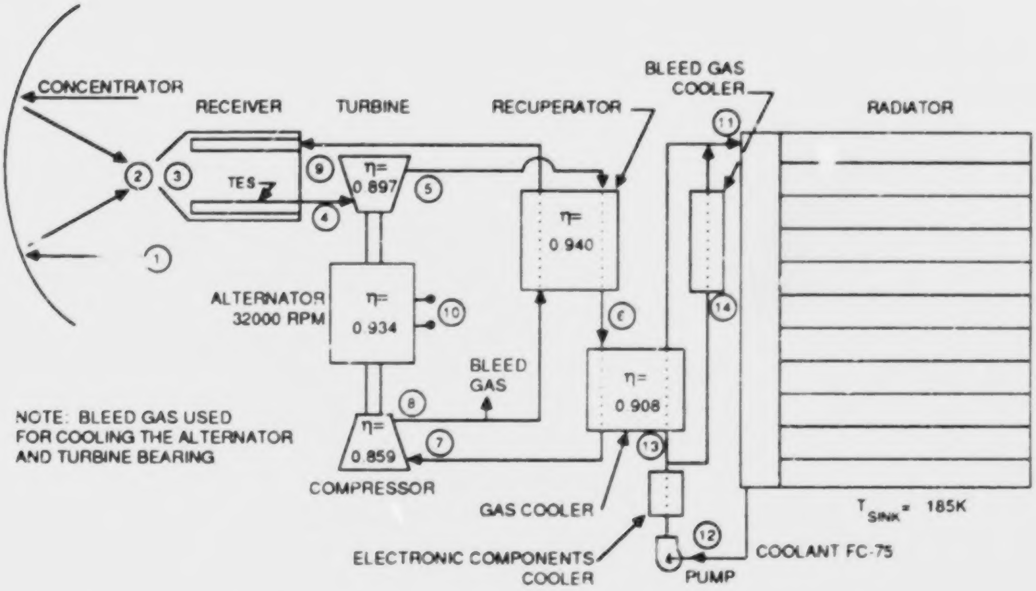
Notes:
1. Included with receiver/TES weight.
2. A single radiator is used for Brayton for combined PCU and electronic components cooler waste heat loads.

Table 6.4.1-4. Solar Dynamic Power Systems Design Data Comparison

| Parameter | 35 kWe | | | 7 kWe | | |
|---|---|---|---|---|---|---|
| | Brayton | Heat Pipe Stirling | Pumped Loop Stirling | Brayton | Heat Pipe Stirling | Pumped Loop Stirling |
| PCU temperature ratio | 3.62 | 2.70 | 2.73 | 3.81 | 2.90 | 2.93 |
| Turbine/engine hot temperature, K$^{(1)}$ | 1086 | 1033 | 1033 | 1086 | 1045 | 1045 |
| Compressor/engine cold temperature, K$^{(1)}$ | 300 | 383 | 383 | 285 | 356 | 357 |
| Geometric concentration ratio$^{(2)}$ | 2000 | 2000 | 2000 | 2000 | 2000 | 2000 |
| Receiver aperture diameter, m | 0.340 | 0.314 | 0.314 | 0.148 | 0.141 | 0.143 |
| Concentrator gross aperture area, m$^{2}$ $^{(3)}$ | 196.3 | 168.1 | 168.3 | 37.44 | 32.97 | 33.73 |
| Block/shadow area, m$^{2}$ | 15.0 | 13.0 | 13.0 | 2.80 | 2.55 | 2.55 |
| Concentrator reflectivity | 0.90 | 0.90 | 0.90 | 0.93 | 0.93 | 0.93 |
| Solar insolation (gross), kWt | 259.7 | 222.3 | 222.6 | 49.53 | 43.62 | 44.62 |
| Reflected to receiver, kWt | 201.9 | 172.6 | 172.8 | 39.84 | 35.00 | 35.87 |
| Receiver intercept loss, kWt | 2.3 | 2.0 | 2.0 | 0.46 | 0.41 | 0.41 |
| Receiver reflective loss, kWt | 1.0 | 3.1 | 4.2 | 0.14 | 0.48 | 0.56 |
| Receiver net solar, kWt | 198.6 | 167.5 | 166.6 | 39.24 | 34.11 | 34.90 |
| Solar multiple$^{(4)}$ | 1.607 | 1.607 | 1.607 | 1.467 | 1.467 | 1.467 |
| Rec. net solar ÷ solar multiple, kWt | 123.5 | 104.2 | 103.7 | 26.75 | 23.25 | 23.79 |
| Reradiation loss, kWt | 9.2 | 7.5 | 6.7 | 1.76 | 1.53 | 1.39 |
| Conduction loss (including TES), kWt | 3.8 | 2.2 | 1.8 | 1.02 | 0.68 | 0.73 |
| Net from receiver to PCU, kWt | 110.5 | 94.5 | 95.2 | 23.97 | 21.04 | 21.67 |
| PCU alternator output, kWe | 39.4 | 39.7 | 40.3 | 8.21 | 8.59 | 8.91 |
| Radiator pump power, kWt | 0.2 | 0.6 | 0.6 | 0.05 | 0.47 | 0.47 |
| PCU waste heat, kWt$^{(5)}$ | 71.3 | 55.4 | 55.4 | 15.80 | 12.92 | 13.24 |
| Frequency converter output, kWe | 36.6 | 36.9 | 37.5 | 7.64 | 7.99 | 8.28 |
| Parasitic power, kWe | 1.6 | 1.9 | 2.5 | 0.64 | 0.99 | 1.28 |
| Net power, kWe | 35.0 | 35.0 | 35.0 | 7.00 | 7.00 | 7.00 |
| Concentrator efficiency (gross)$^{(6)}$ | 0.777 | 0.776 | 0.776 | 0.804 | 0.802 | 0.804 |
| Receiver interception efficiency$^{(7)}$ | 0.894 | 0.951 | 0.964 | 0.985 | 0.975 | 0.973 |
| Receiver/TES efficiency | 0.895 | 0.907 | 0.918 | 0.896 | 0.905 | 0.911 |
| PCU plus alternator efficiency | 0.356 | 0.420 | 0.424 | 0.342 | 0.408 | 0.411 |
| Frequency converter efficiency | 0.930 | 0.930 | 0.930 | 0.930 | 0.930 | 0.930 |
| Efficiency - solar (gross) to net power$^{(8)}$ | 0.217 | 0.253 | 0.253 | 0.207 | 0.235 | 0.230 |
| Alternator output frequency, kHz | 0.533 | 0.095 | 0.095 | 0.800 | 0.150 | 0.150 |
| Converter output frequency, kHz | 20 | 20 | 20 | 20 | 20 | 20 |
| Electronic components cooler (ECC), kWt | 3.5 | 3.5 | 3.5 | 1.0 | 1.0 | 1.0 |
| PCU + ECC radiator radiant area, m$^{2}$ $^{(9)}$ | 211.2 | 113.4 + 23.2 | 113.2 + 23.2 | 58.78 | 33.8 + 7.0 | 34.6 + 7.0 |
| PCU + ECC radiator sail area, m$^{2}$ $^{(9)}$ | 109.8 | 59.0 + 12.2 | 58.9 + 12.2 | 31.74 | 18.3 + 3.9 | 18.7 + 3.9 |
| Power system weight, kg | 5067 | 3939 | 4485 | 1418 | 1158 | 1341 |

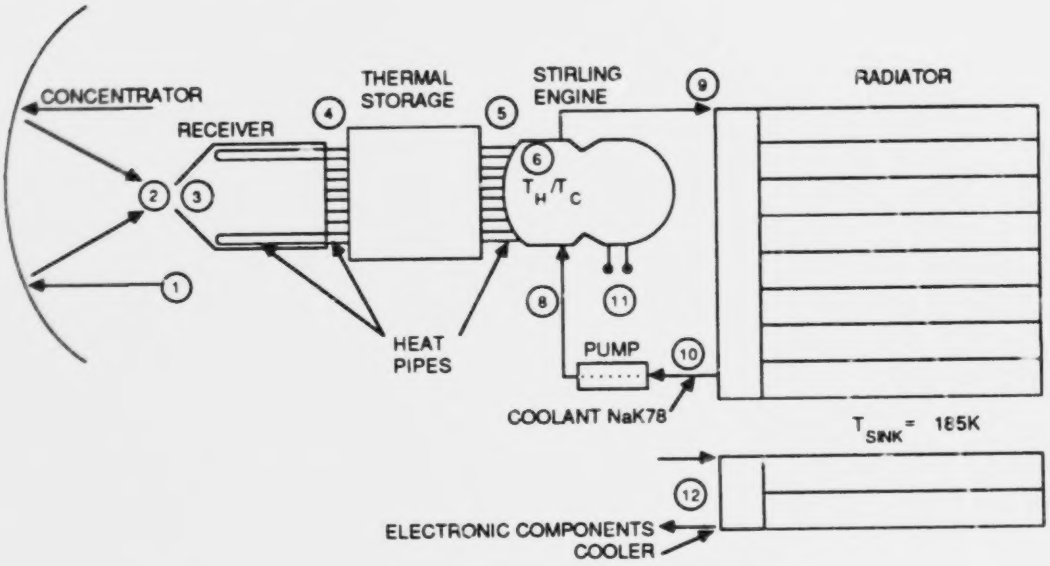
Notes:

1. CBC turbine inlet and compressor inlet temperatures , or Stirling cycle engine hot and cold temperatures
2. Based on concentrator net aperture area, which includes an increase due to hex segment packing factor (reflective facet area ÷ hex area = 0.935)
3. Includes blockage and shadow area
4. Orbit period ÷ shortest sun interval for the orbit
5. Includes EM pump power for Stirling cycles: 0.6 kW at 35 kWe and 0.5 kW at 7 kWe power levels
6. Reflected to receiver ÷ solar insolation (gross)
7. Receiver net solar ÷ reflected to receiver
8. Net power ÷ (solar insolation [gross] ÷ solar multiple)
9. A single radiator is used for Brayton for combined PCU and ECC waste heat loads


CONCENTRATOR
RECEIVER
TURBINE
RECUPERATOR
BLEED GAS COOLER
RADIATOR
TES
η= 0.897
η= 0.940
ALTERNATOR 32000 RPM
η= 0.934
BLEED GAS
η= 0.908
NOTE: BLEED GAS USED FOR COOLING THE ALTERNATOR AND TURBINE BEARING
η= 0.859
COMPRESSOR
GAS COOLER
ELECTRONIC COMPONENTS COOLER
PUMP
COOLANT FC-75
$T_{SINK}$ = 185K
①
②
③
④
⑤
⑥
⑦
⑧
⑨
⑩
⑪
⑫
⑬
⑭

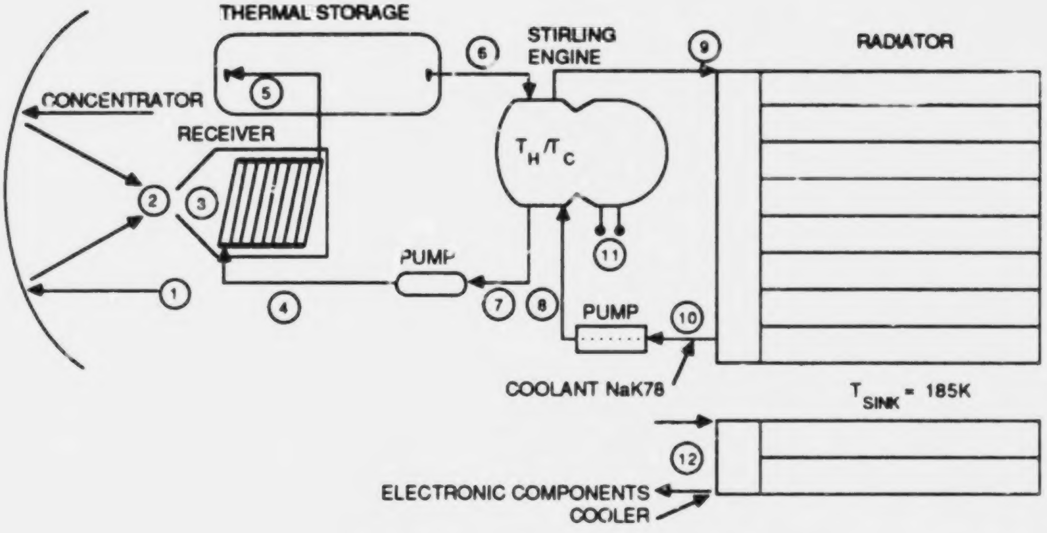
| State Point | Power.kW | Temperature, K | Pressure, kPa | Flowrate, kg/sec |
|---|---|---|---|---|
| 1 | 259.7 | | | |
| 2 | 201.9 | | | |
| 3 | 198.6 | | | |
| 4 | | 1086 | 288 | 0.923 |
| 5 | | 890 | 164 | 0.923 |
| 6 | | 437 | 162 | 0.923 |
| 7 | | 300 | 160 | 0.947 |
| 8 | | 400 | 301 | 0.947 |
| 9 | | 858 | 299 | 0.923 |
| 10 | 39.4 | | | |
| 11 | | 398 | | 0.624 |
| 12 | | 280 | | 0.624 |
| 13 | | 286 | | 0.548 |
| 14 | | 286 | | 0.076 |
| 4-5 | 95.5 | | | |
| 5-6 | 222.4 | | | |
| 6-7 | 68.5 | | | |
| 7-8 | 50.3 | | | |
| 8-9 | 222.4 | | | |
| 9-4 | 110.5 | | | |
| 11-12 | 74.8 | | | |
| 12-13 | 3.5 | | | |
| 13-11 | 71.3 | | | |
| 14-11 | 2.8 | | | |

Figure 6.4.1-1. 35 kWe Brayton Cycle State Point Diagram


CONCENTRATOR
RECEIVER
THERMAL STORAGE
STIRLING ENGINE
RADIATOR
①
②
③
④
⑤
⑥
$T_H/T_C$
⑧
⑨
⑩
⑪
⑫
HEAT PIPES
PUMP
COOLANT NaK78
$T_{SINK}$ = 185K
ELECTRONIC COMPONENTS COOLER

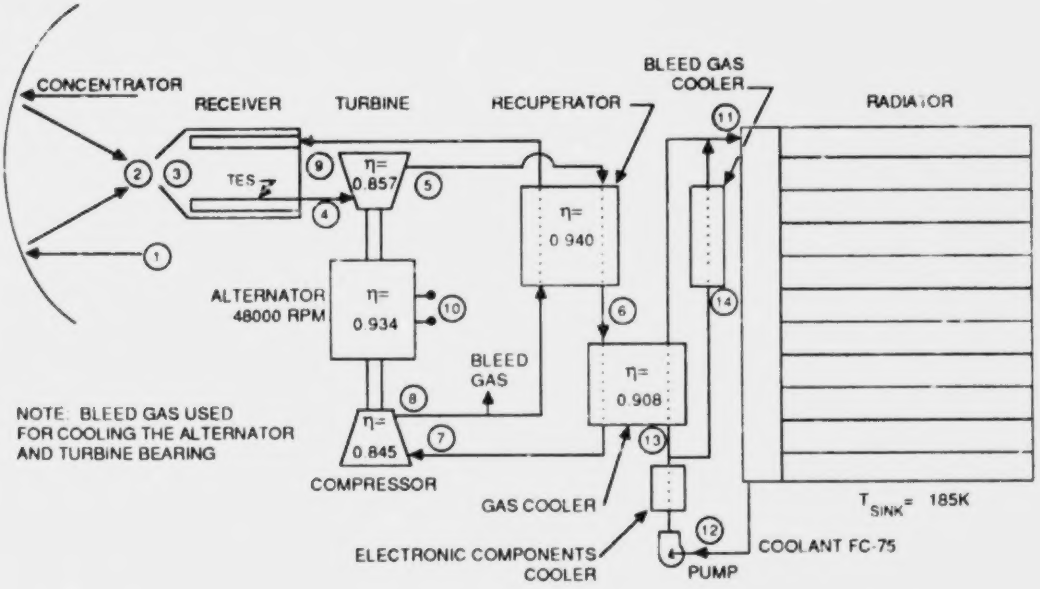
| State Point | Power, kW | Temperature, K | Flowrate, kg/sec |
|---|---|---|---|
| 1 | 222.3 | | |
| 2 | 172.6 | | |
| 3 | 167.5 | | |
| 4 | 159.1 | | |
| 5 | 94.5 | | |
| 6 | | 1033 | |
| 8 | | 342 | 1.49 |
| 9 | | 382 | 1.49 |
| 10 | | 341 | 1.49 |
| 11 | 39.7 | | |
| 12 | 3.5 | | |
| 8-9 | 54.8 | | |
| 9-10 | 55.4 | | |
| 10-8 | 0.6 | | |

Figure 6.4.1-2. 35 kWe Heat Pipe Stirling Cycle State Point Diagram


THERMAL STORAGE
STIRLING ENGINE
RADIATOR
CONCENTRATOR
RECEIVER
$T_H/T_C$
PUMP
PUMP
COOLANT NaK78
$T_{SINK}$ = 185K
ELECTRONIC COMPONENTS COOLER
①
②
③
④
⑤
⑥
⑦
⑧
⑨
⑩
⑪
⑫

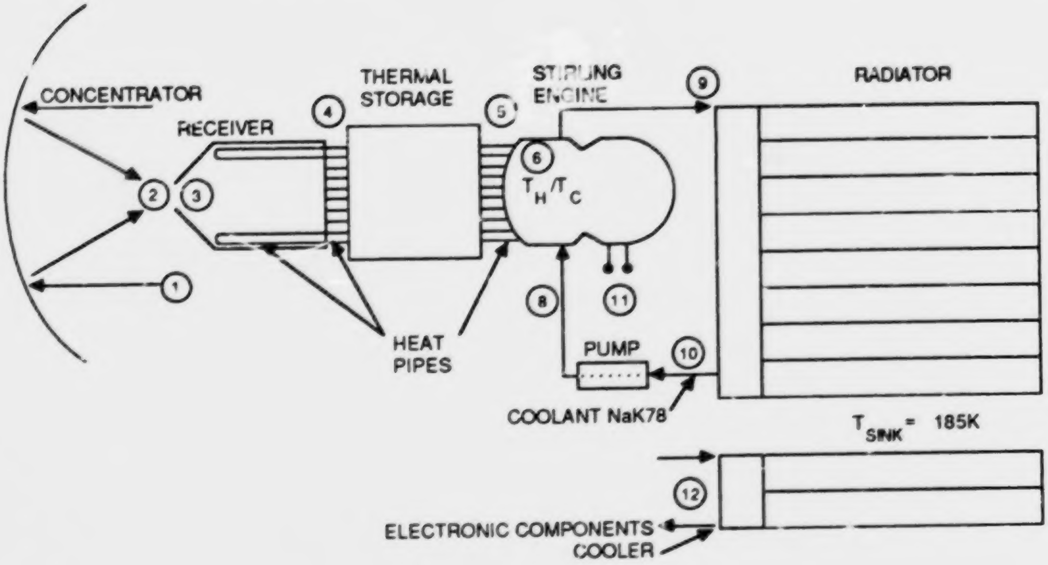
| State Point | Power, kW | Temperature, K | Flowrate, kg/sec |
|---|---|---|---|
| 1 | 222.6 | | |
| 2 | 172.8 | | |
| 3 | 166.6 | | |
| 4 | | 1031 | 2.65 |
| 5 | | 1098 | 2.65 |
| 6 | | 1071 | 2.65 |
| 7 | | 1031 | 2.65 |
| 8 | | 342 | 1.49 |
| 9 | | 382 | 1.49 |
| 10 | | 341 | 1.49 |
| 11 | 40.3 | | |
| 12 | 3.5 | | |
| 4-5 | 159.3 | | |
| 5-6 | 64.1 | | |
| 6-7 | 95.2 | | |
| 7-4 | 0.1 | | |
| 8-9 | 54.8 | | |
| 9-10 | 55.4 | | |
| 10-8 | 0.6 | | |

Figure 6.4.1-3. 35 kWe Pumped Loop Stirling Cycle State Point Diagram


CONCENTRATOR
RECEIVER
TURBINE
RECUPERATOR
BLEED GAS COOLER
RADIATOR
TES
η= 0.857
η= 0.940
η= 0.934
η= 0.908
η= 0.845
ALTERNATOR 48000 RPM
BLEED GAS
NOTE: BLEED GAS USED FOR COOLING THE ALTERNATOR AND TURBINE BEARING
COMPRESSOR
GAS COOLER
ELECTRONIC COMPONENTS COOLER
PUMP
COOLANT FC-75
$T_{SINK}$ = 185K
1
2
3
4
5
6
7
8
9
10
11
12
13
14

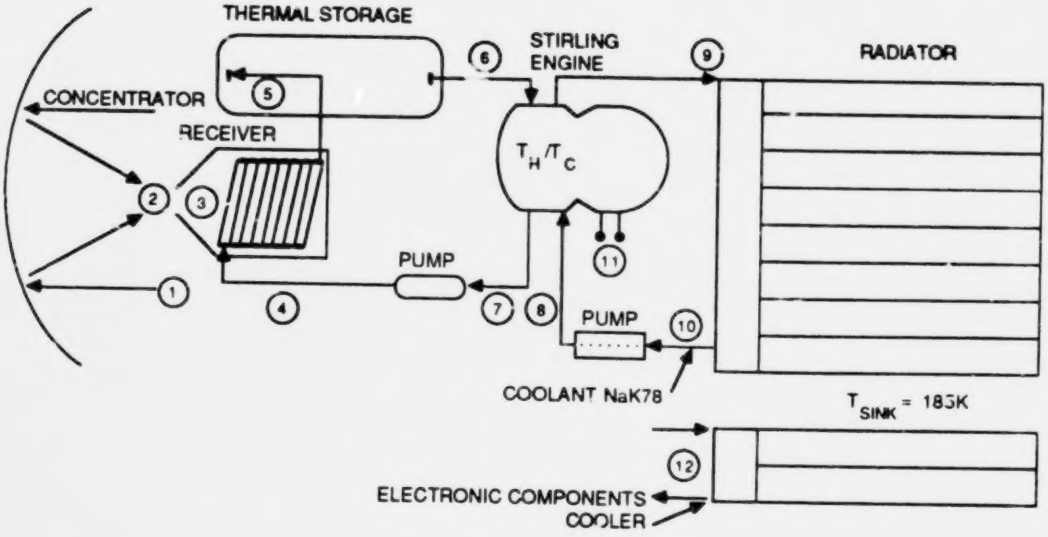
| State Point | Power.kW | Temperature, K | Pressure, kPa | Flowrate, kg/sec |
|---|---|---|---|---|
| 1 | 49.5 | | | |
| 2 | 39.8 | | | |
| 3 | 39.2 | | | |
| 4 | | 1086 | 288 | 0.252 |
| 5 | | 902 | 164 | 0.252 |
| 6 | | 420 | 162 | 0.252 |
| 7 | | 285 | 160 | 0.258 |
| 8 | | 382 | 301 | 0.258 |
| 9 | | 866 | 299 | 0.252 |
| 10 | 8.2 | | | |
| 11 | | 382 | | 0.144 |
| 12 | | 263 | | 0.144 |
| 13 | | 271 | | 0.127 |
| 14 | | 271 | | 0.017 |
| 4-5 | 20.4 | | | |
| 5-6 | 53.0 | | | |
| 6-7 | 15.2 | | | |
| 7-8 | 11.0 | | | |
| 8-9 | 53.0 | | | |
| 9-4 | 24.0 | | | |
| 11-12 | 16.8 | | | |
| 12-13 | 1.0 | | | |
| 13-11 | 15.8 | | | |
| 14-11 | 0.6 | | | |

Figure 6.4.1-4. 7 kWe Brayton Cycle State Point Diagram


CONCENTRATOR
RECEIVER
THERMAL STORAGE
STIRLING ENGINE
RADIATOR
①
②
③
④
⑤
⑥
$T_H/T_C$
⑧
⑨
⑩
⑪
⑫
HEAT PIPES
PUMP
COOLANT NaK78
$T_{SINK}$ = 185K
ELECTRONIC COMPONENTS COOLER

| State Point | Power, kW | Temperature, K | Flowrate, kg/sec |
|---|---|---|---|
| 1 | 43.6 | | |
| 2 | 35.0 | | |
| 3 | 34.1 | | |
| 4 | 32.3 | | |
| 5 | 21.0 | | |
| 6 | | 1033 | |
| 8 | | 327 | 0.552 |
| 9 | | 351 | 0.552 |
| 10 | | 326 | 0.552 |
| 11 | 8.0 | | |
| 12 | 1.0 | | |
| 8-9 | 12.4 | | |
| 9-10 | 12.9 | | |
| 10-8 | 0.5 | | |

Figure 6.4.1-5. 7 kWe Heat Pipe Stirling Cycle State Point Diagram


THERMAL STORAGE
STIRLING ENGINE
RADIATOR
CONCENTRATOR
RECEIVER
$T_H/T_C$
PUMP
PUMP
COOLANT NaK78
$T_{SINK}$ = 185K
ELECTRONIC COMPONENTS COOLER
①
②
③
④
⑤
⑥
⑦
⑧
⑨
⑩
⑪
⑫

| State Point | Power, kW | Temperature, K | Flowrate, kg/sec |
|---|---|---|---|
| 1 | 44.6 | | |
| 2 | 35.9 | | |
| 3 | 34.9 | | |
| 4 | | 1036 | 0.603 |
| 5 | | 1097 | 0.603 |
| 6 | | 1076 | 0.603 |
| 7 | | 1036 | 0.603 |
| 8 | | 327 | 0.565 |
| 9 | | 351 | 0.565 |
| 10 | | 326 | 0.565 |
| 11 | 8.3 | | |
| 12 | 1.0 | | |
| 4-5 | 33.3 | | |
| 5-6 | 11.6 | | |
| 6-7 | 21.7 | | |
| 7-4 | 0.1 | | |
| 8-9 | 12.7 | | |
| 9-10 | 13.2 | | |
| 10-8 | 0.5 | | |

Figure 6.4.1-6. 7 kWe Pumped Loop Stirling Cycle State Point Diagram

### 6.4.2 35 kWe Closed Brayton Cycle Design Trades

This section presents the effect of compressor inlet temperature, compressor inlet pressure, recuperator effectiveness, pressure ratio factor, speed, and TES material selection as these parameters effect power system performance (weight and area). At the outset of the trade study, a number of these design parameters were chosen to be equal to the Space Station 25 kWe CBC values, due to the similarity of the two designs from an alternator power output basis (32.1 kWe versus 39.4 kWe for this study). As the trade study progressed, several of the parameters were adjusted somewhat so as to be nearer minimum power system weight conditions, with the exception of rotor speed. The process of minimum weight optimization was not pursued to completion, as to do so was beyond the scope of the study.

Figure 6.4.2-1 presents system weight as a function of compressor inlet temperature for the three TES materials. The choice of LiF resulted in lower system weight than either NaF or $Mg_2Si$. The nominal compressor inlet temperature was chosen as 300K based on limitations of the radiator temperature under conditions maximum solar energy input.

Figure 6.4.2-2 presents the system area curves for the case of LiF TES material, as is the case for the remaining figures in this section. Concentrator area is maximum effective aperture area (required aperture plus shading and increased by the concentrator surface packing factor). Radiator sail area is planform area including heat exchanger boom. The equivalent area relates to power system drag area. The equivalent area equals the concentrator area plus half of the radiator sail area, the combination approximating the fact that the radiator drag coefficient is less than half that of the concentrator for the Space Station (ref. 1). Equivalent area varies only slightly with compressor inlet temperature.

Figures 6.4.2-3 and 6.4.2-4 present system weight as a function of compressor inlet temperature and compressor pressure ratio. The nominal compressor pressure ratio was selected as 1.88, slightly less than the Space Station CBC design value, and represents near-minimum weight for this design.

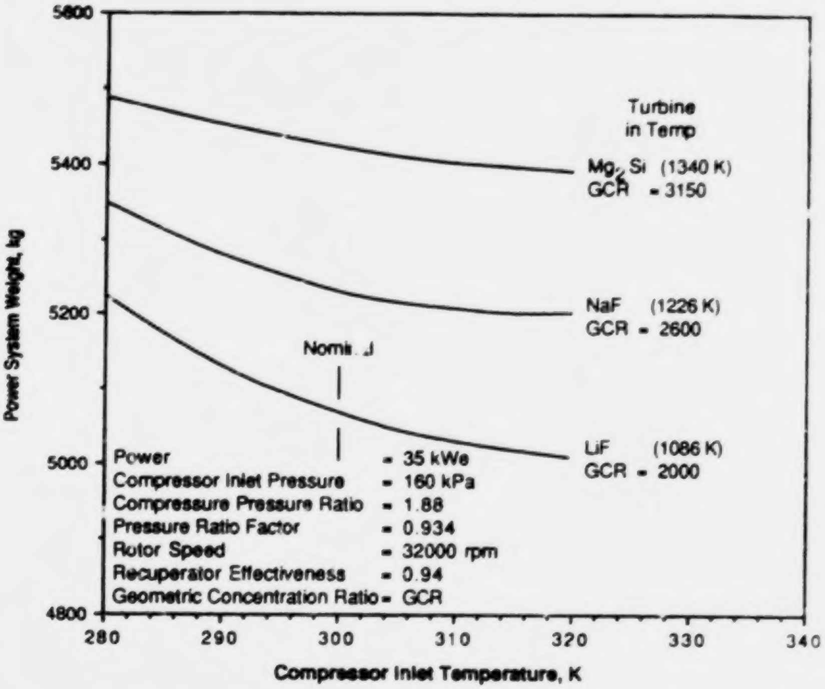


Figure 6.4.2-1. 35 kWe CBC Weight vs TES Material and Compressor Temperature

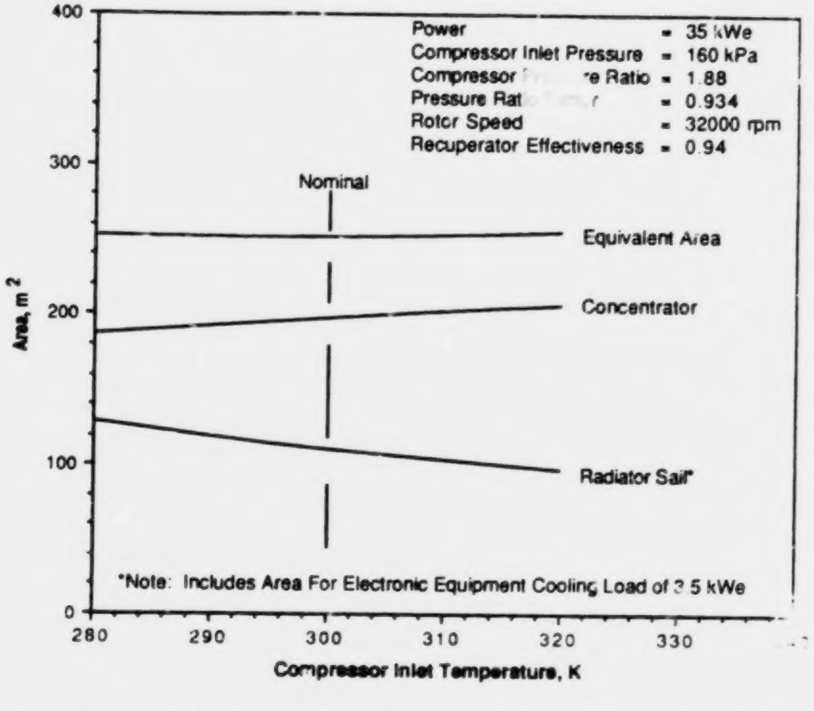


Figure 6.4.2-2. 35 kWe CBC Exposed Area vs Compressor Temperature

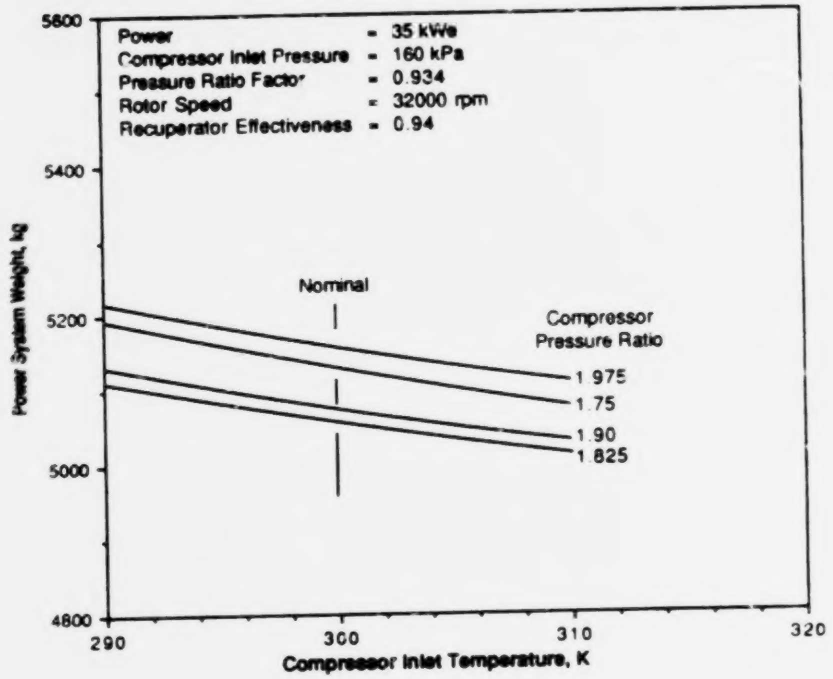


Figure 6.4.2-3. 35 kWe CBC Weight vs Compressor Pressure Ratio and Temperature

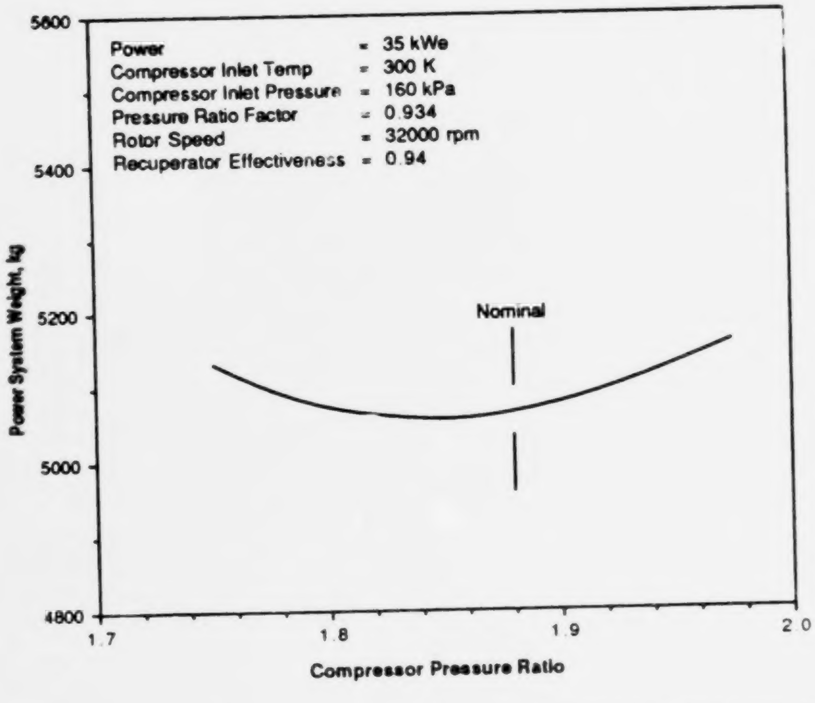


Figure 6.4.2-4. 35 kWe CBC Weight vs Compressor Pressure Ratio

Figure 6.4.2-5 presents system weight as a function of compressor inlet pressure. The nominal pressure was selected as 160 kPa, somewhat less than the Space Station CBC design value of 186 kPa, and represents near minimal weight for this design.

Figure 6.4.2-6 and 6.4.2-7 present two interrelated CBC engine design parameters, recuperator effectiveness and the engine pressure ratio factor (the ratio of turbine pressure ratio to compressor pressure ratio). Increases in recuperator effectiveness tend to increase recuperator pressure drop, which would cause reduction in pressure ratio factor. An analytical relationship between the two parameters was not developed for this study. The nominal values are equal to the Space Station CBC design values.

Figure 6.4.2-8 presents system weight as a function of rotor speed. The nominal speed of 32,000 rpm was chosen, equal to the Space Station CBC design value. This resulted in a system weight approximately 1.8% above the minimum weight of about 4975 kg shown in the figure. Further optimization of this and other parameters would be appropriate, and could potentially lead to a 35 kWe CBC power system weight of ≤4950 kg. This weight would still be significantly higher than for the 35 kWe heat pipe Stirling design.

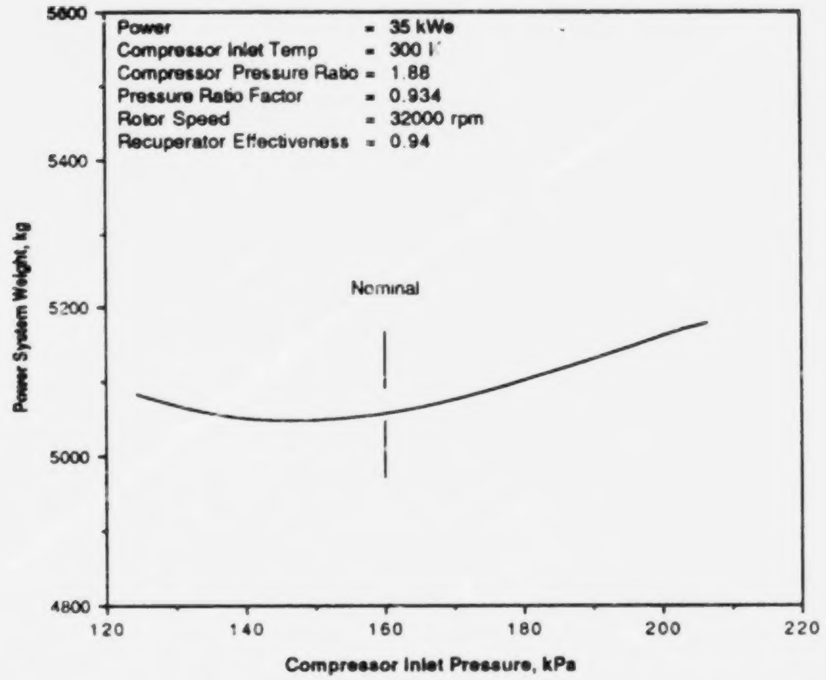


Figure 6.4.2-5. 35 kWe CBC Weight vs Compressor Inlet Pressure

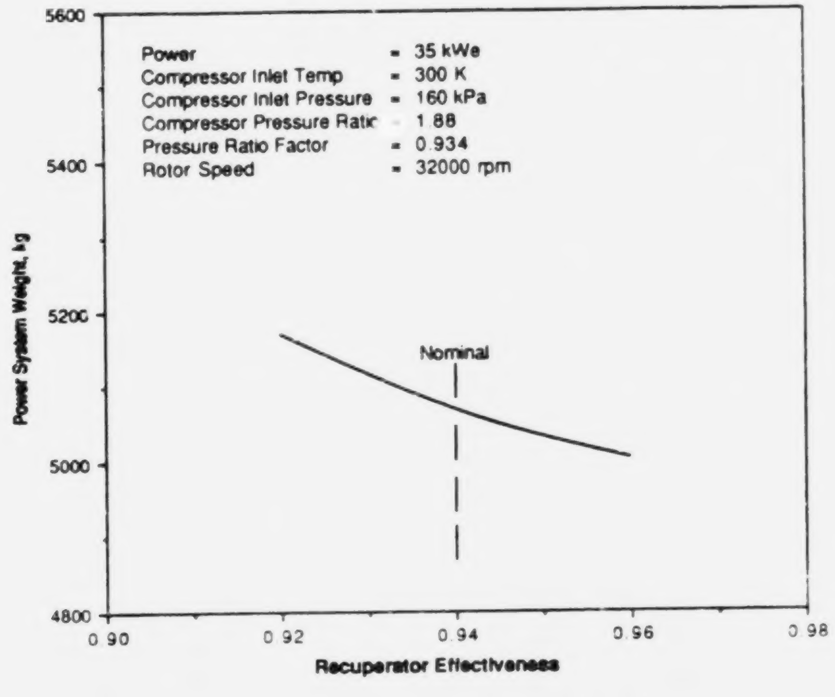


Figure 6.4.2-6. 35 kWe CBC Weight vs Recuperator Effectiveness

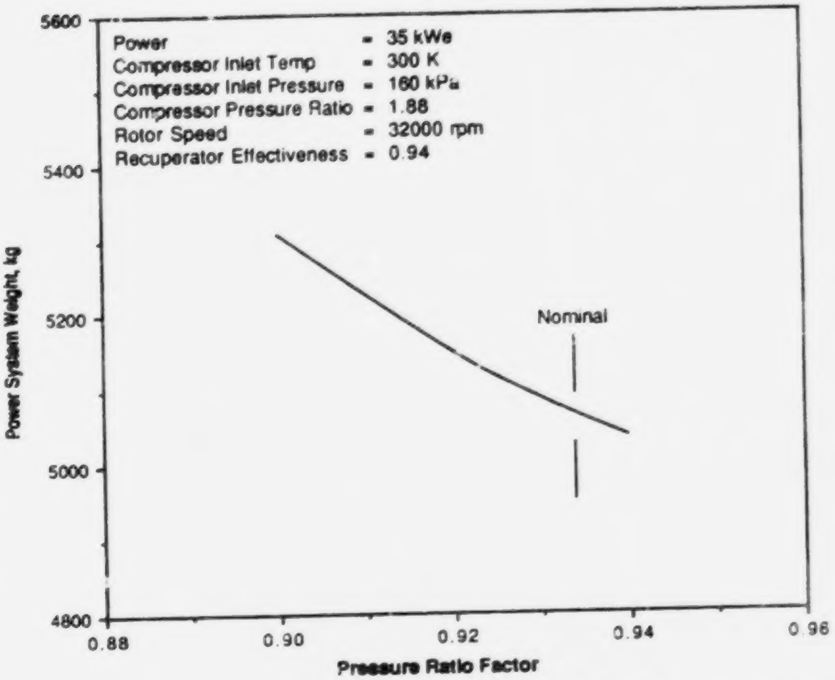


Figure 6.4.2-7. 35 kWe CBC Weight vs Pressure Ratio Factor

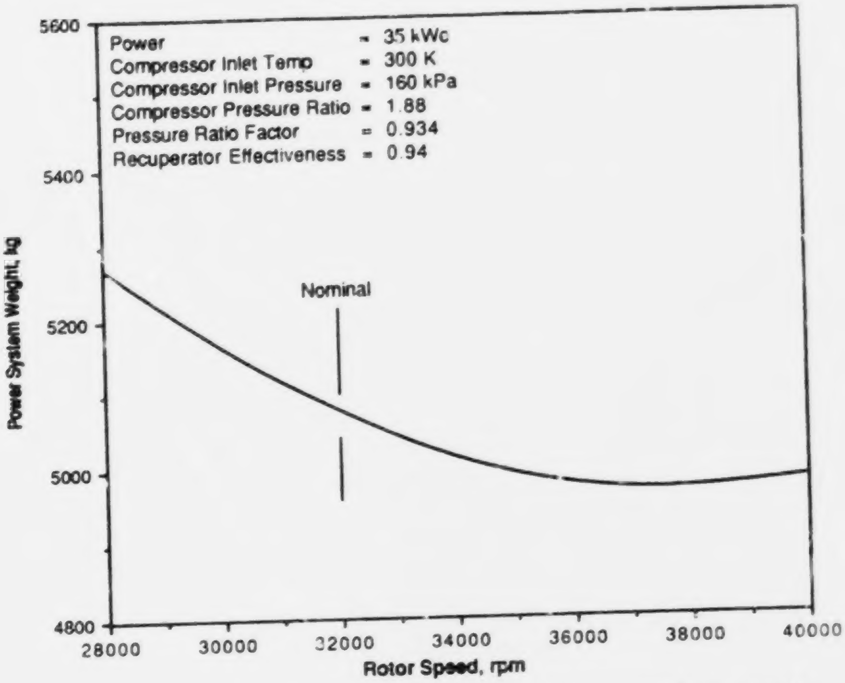


Figure 6.4.2-8. 35 kWe CBC Weight vs Rotor Speed

### 6.4.3 35 kWe Heat Pipe Stirling Cycle Design Trades

This section presents the effect of engine cold temperature, engine specific weight, solar concentration ratio, concentrator pointing and surface slope error, and TES material selection as these parameters effect power system performance (weight and area).

Figure 6.4.3-1 presents system weight as a function of engine cold temperature for the three TES materials. Appropriate values for engine hot temperature and geometric concentration ratio are indicated. As stated previously, material density/weight were not altered when switching from superalloys to refractory alloys. Minimum weight for LiF and for NaF are nearly the same, with weight of the $Mg_2Si$ design being only 1.5% higher. The temperatures for the minimum weight points of the curves are nearly the same, ranging between 395K and 415K. Limitation of maximum radiator temperature for the conditions of maximum solar energy input resulted in the limitation of nominal $T_C$ to no more than about 383K for the design conditions of minimum solar input.

Figures 6.4.3-2 and 6.4.3-3 present the system area curves corresponding to the prior figure. Figure 6.4.3-2 presents concentrator maximum effective aperture area (required aperture plus shading) and radiator sail area (planform area including heat exchanger boom), for the three TES materials. Figure 6.4.3-3 presents concentrator area, radiator area, and an equivalent area which relates to drag area (see section 6.4.2), for LiF TES. A near-minimum of equivalent area occurs for the design $T_C$ of 383K.

Figure 6.4.3-4 presents system weight as a function of geometric concentration ratio for the three TES materials. The nominal concentration ratios were chosen somewhat arbitrarily corresponding with the point at which system weight was about 2/3% greater than the minimum weight, which in turn resulted in concentration ratios approximately 2/3 of the minimum weight values. The primary reason for this selection approach was that the analysis did not include any effect of change in concentrator facet design with possible increased concentration ratio, and the consequential increase in

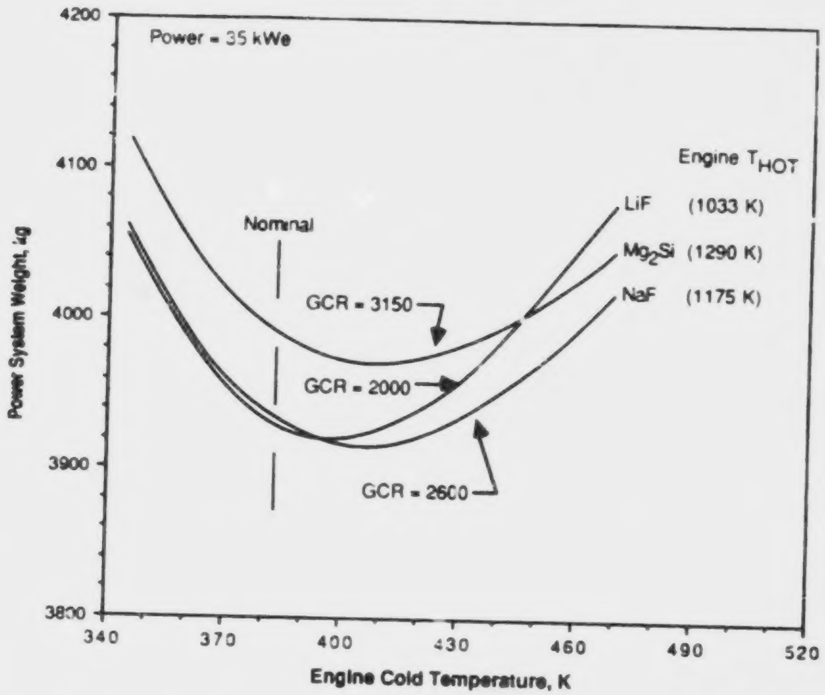


Figure 6.4.3-1. 35 kWe HP Stirling Weight vs TES Material and Cold Temperature

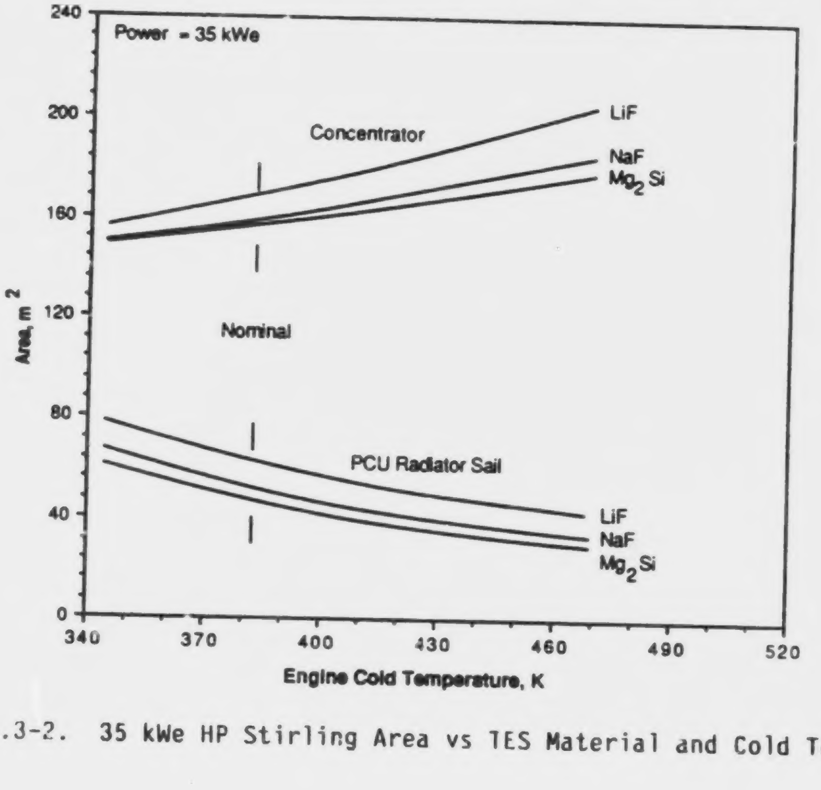


Figure 6.4.3-2. 35 kWe HP Stirling Area vs TES Material and Cold Temperature

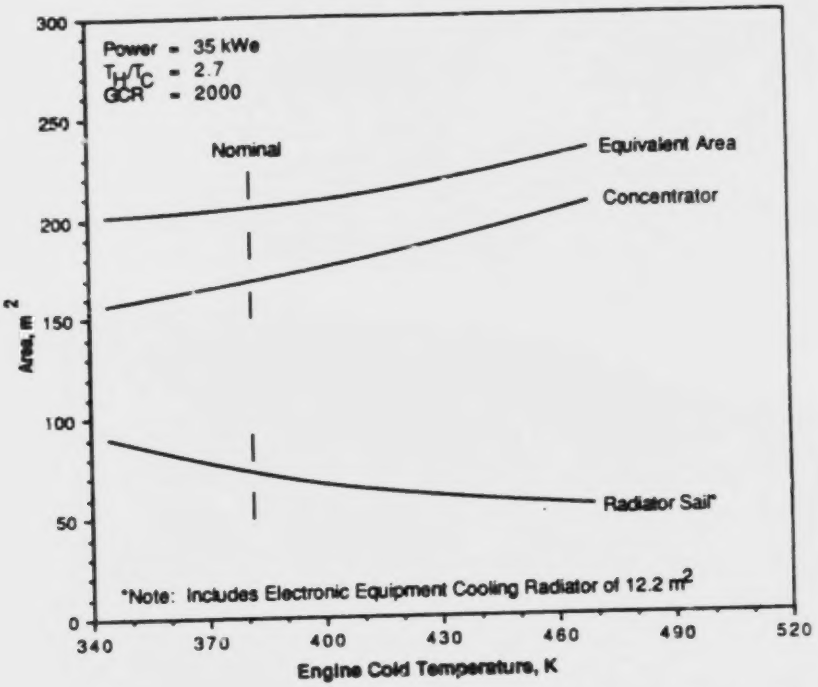


Figure 6.4.3-3. 35 kWe HP Stirling Exposed Area vs Cold Temperature

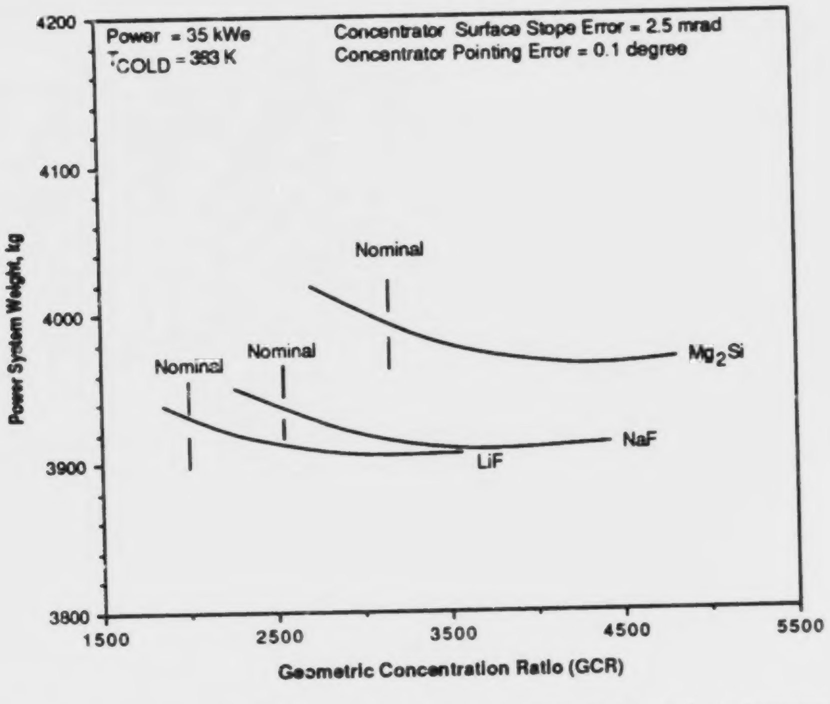


Figure 6.4.3-4. 35 kWe HP Stirling Weight vs TES Material & Concentration Ratio

weight, reduction in packing factor, etc. The nominal concentration ratios chosen were:

| TES Material | Geometric Concentration Ratio |
|---|---|
| LiF | 2000 |
| NaF | 2600 |
| $Mg_2Si$ | 3150 |

Figure 6.4.3-5 presents the results of system weight as a function of engine specific weight for the 35 kWe power level and LiF TES. The engine efficiency versus specific weight was chosen from figure 6.2.1.4-2 for $T_H/T_C = 2.7$. The nominal engine specific weight of 8.0 (increased by a factor of 1.067 for oversizing the alternator for excess power) results in essentially minimum system weight.

Figure 6.4.3-6 presents system weight as a function of concentrator pointing error and surface slope error, for the nominal concentration ratio of 2000. The nominal slope error of 2.5 mrad results in 0.2% higher system weight than for 1 mrad, for the nominal pointing error of 0.1 degree. Little additional weight increase is indicated by increasing pointing error to 0.2 degree or more, for the 2.5 mrad slope error.

Figures 6.4.3-7 and 6.4.3-8 present system weight as a function of concentrator pointing error and concentration ratio, for pointing errors of 0.1 and 0.2 degrees. Figure 6.4.3-8 confirms that an increase in nominal pointing error to 0.2 degrees would be appropriate.

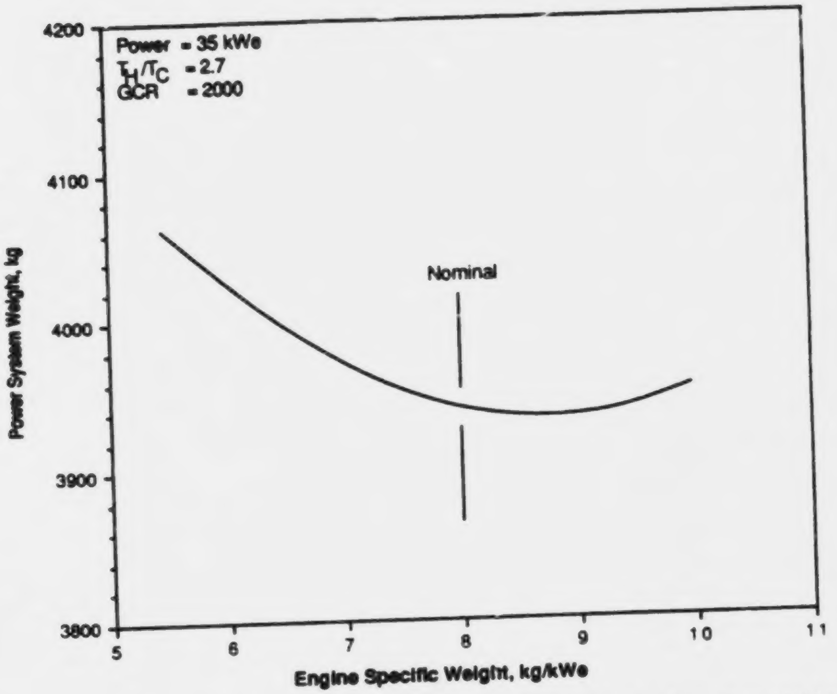


Figure 6.4.3-5. 35 kWe HP Stirling Weight vs Engine Specific Weight

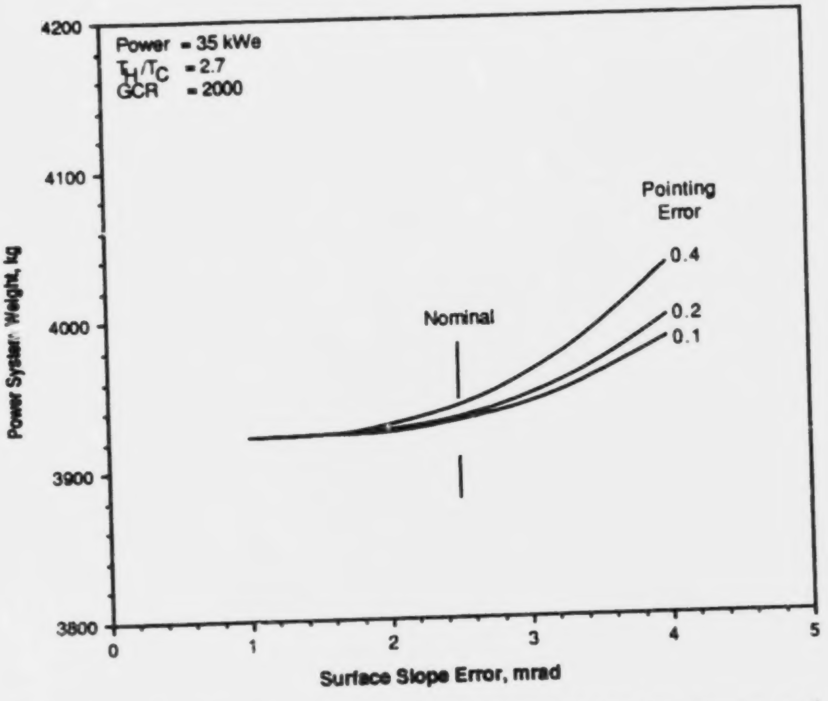


Figure 6.4.3-6. 35 kWe HP Stirling Weight vs Pointing Error and Slope Error

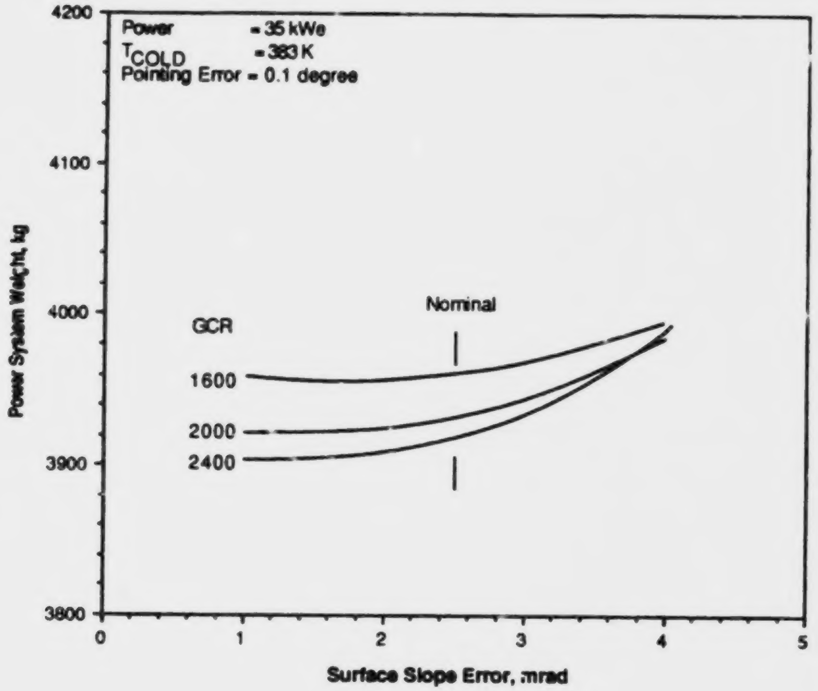


Figure 6.4.3-7. 35 kWe HP Stirling Weight vs Concentration Ratio and Slope Error (Pointing Error 0.1°)

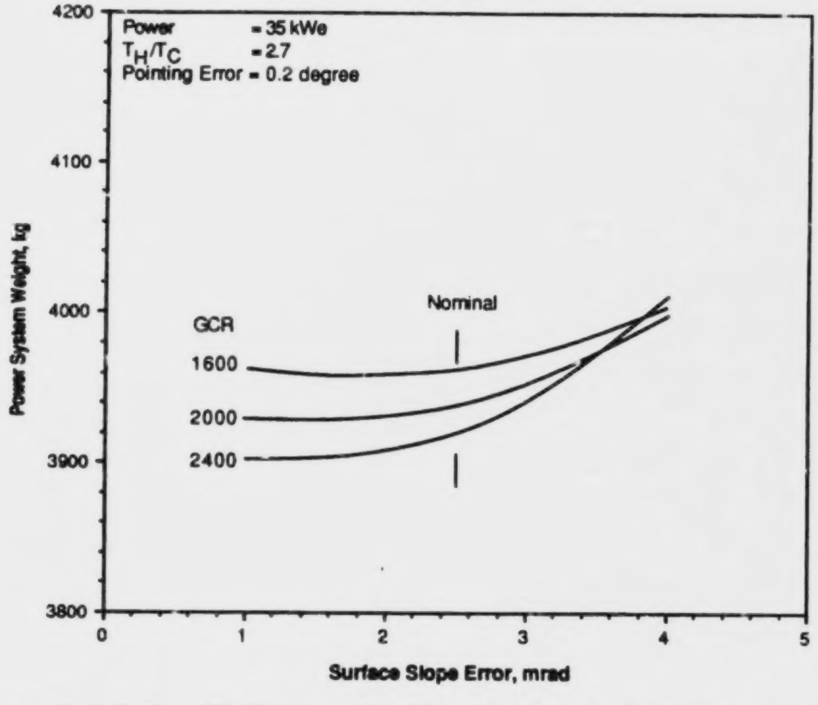


Figure 6.4.3-8. 35 kWe HP Stirling Weight vs Concentration Ratio and Slope Error (Pointing Error 0.2°)

### 6.4.4 Splined Radial Panel Concentrator Trade

A comparison was made between the use of the truss hex concentrator versus the splined radial panel (SRP) concentrator. Equations used to determine the weight of each type of concentrator were presented in section 6.3.2.1. The concentrator weight reduction for the 35 kWe application was more than 60% for the SRP design.

Figure 6.4.4-1 presents CBC power system weight as a function of compressor inlet temperature, and figure 6.4.4-2 presents heat pipe Stirling power system weight as a function of engine cold temperature $T_C$. In both cases, substantial system weight reductions would be realized with the SRP concentrator. From a system weight optimization standpoint, choice of a light weight concentrator tends to allow for a larger concentrator area and reduced radiator area at the minimum weight condition. The smaller radiator would be somewhat warmer, with correspondingly higher cold end temperatures for either engine type. The radiator methanol temperature limitation for the case of maximum solar energy input would not permit selection of the minimum weight condition for either the truss hex concentrator or SRP concentrator; however, this would be a second order effect as compared to the differences in weight between the two concentrator designs.

The SRP concentrator, shown in figure 6.2.1.1-3, is a symmetric parabolic configuration, whereas the truss hex concentrator is an offset configuration. Placement of the balance of the power system components and location of the host spacecraft are much more easily accommodated with the offset design, which is designed so as to avoid shadowing (by other than the concentrator support structure). It is believed that an offset SRP can be designed, but since this has not been attempted, associated weight and area penalties are unknown.

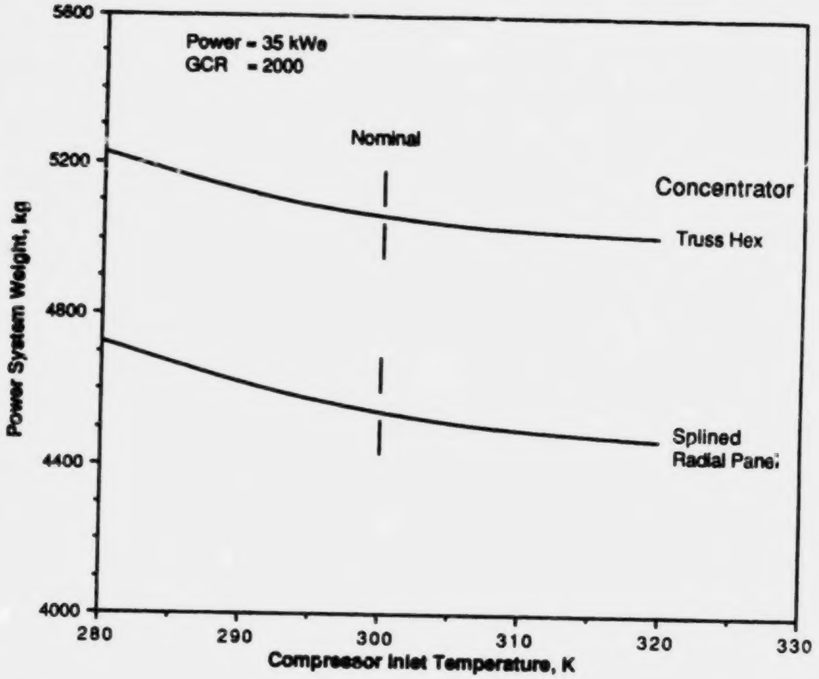


Figure 6.4.4-1. 35 kWe CBC Weight - Effect of Concentrator Type

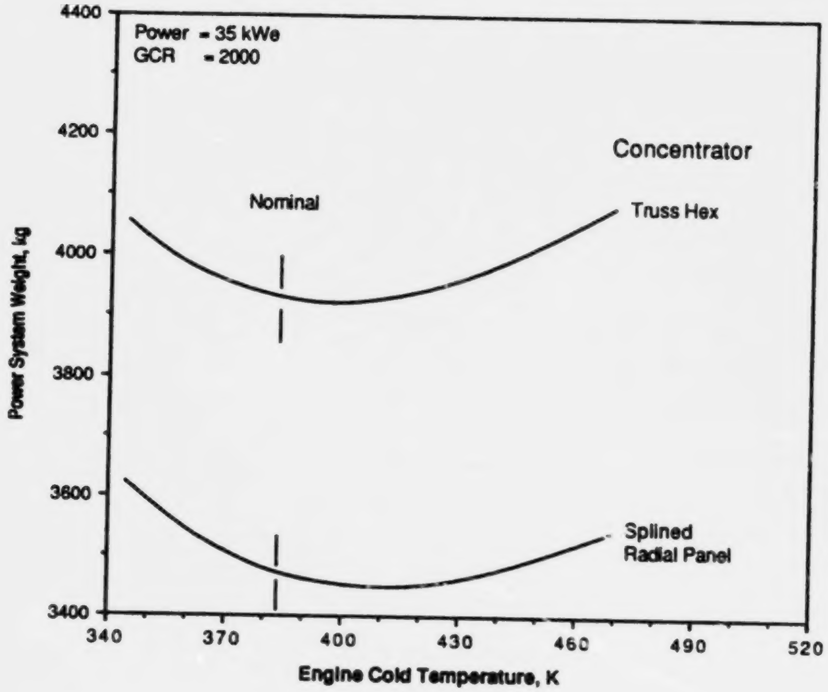


Figure 6.4.4-2. 35 kWe HP Stirling Weight - Effect of Concentrator Type

### 6.4.5 Radiator Panel Heat Pipe Wall Thickness Trade

This section presents the results of a trade of radiator heat pipe wall thickness versus the number of additional heat pipe panels. Heat pipe failure rates due to micrometeoroid and debris hazard were obtained from the work performed for reference 2. Failure rates and panel weights for the titanium/methanol dual-slot heat pipe radiator panels are presented in table 6.4.5-1.

The Space Station analysis presented in reference 2 approached redundancy from a life cycle cost approach, for a serviceable circumstance, which resulted in the choice of 0.178 cm (0.070 in.) titanium pipe wall thickness. The unserviceable circumstance was examined for this study, and the selection criterion chosen was minimum radiator panel weight. The redundancy analysis did not include the reliability effect of the heat exchanger boom; however, the weight effect was included, so as to reflect total radiator weight.

The analysis was performed for an unserviceable spacecraft, for a reliability goal of $\geq$0.995 probability of having the minimum number of panels (based on thermal requirements) in operation at the end of 7 year life. The results are possibly conservative, as no credit was taken for additional protection possibly provided by presence of the fin on the heat pipe (see section 10.6.2). No heat exchanger clamping failure rate was included for the circumstance of the unserviceable spacecraft.

Figure 6.4.5-1 presents total radiator weight as a function of the number of redundant panels, for the case of 21 panels based on thermal duty. The 24-panel nominal radiator design results in near minimum weight. A radiator design with an odd number of panels (i.e. 23 or 25) would actually be heavier than shown on figure 6.4.5-1, due to the double-sided heat exchanger boom having to be long enough for the odd panel, a factor not included in the analysis. Figure 6.4.5-2 presents the heat pipe wall thickness necessary to obtain the required reliability of 0.995. The nominal wall thickness for the 24-panel (3 redundant) radiator design was determined to be 0.152 cm (0.060 in.). The heat pipe nominal inside diameter was 1.905 cm (0.750 in.) for all cases.

Table 6.4.5-1. Heat Pipe Radiator Panel Failure Rates

Titanium-Methanol Dual-Slot Heat Pipe Radiator Panels
25 ft x 1 ft Condenser Section, 2 ft x 1 ft Evaporator Section
2 Condenser Legs, 10 Evaporator Legs, 0.032 in. Fin Thickness

Wall Thickness = 0.030 inch
Panel Failure Rate = 0.0456 per year
Radiator Panel Weight 30.1 lbs

Wall Thickness = 0.040 inch
Panel Failure Rate = 0.0169 per year
Radiator Panel Weight 33.7 lbs

Wall Thickness = 0.050 inch
Panel Failure Rate = 0.0079 per year
Radiator Panel Weight 37.3 lbs

Wall Thickness = 0.060 inch
Panel Failure Rate = 0.0042 per year
Radiator Panel Weight 41.1 lbs

Wall Thickness = 0.070 inch
Panel Failure Rate = 0.0025 per year
Radiator Panel Weight 44.9 lbs

Wall Thickness = 0.080 inch
Panel Failure Rate = 0.0016 per year
Radiator Panel Weight 48.8 lbs

Wall Thickness = 0.090 inch
Panel Failure Rate = 0.0010 per year
Radiator Panel Weight 52.8 lbs

Wall Thickness = 0.100 inch
Panel Failure Rate = 0.0007 per year
Radiator Panel Weight 56.9 lbs

Wall Thickness = 0.110 inch
Panel Failure Rate = 0.0005 per year
Radiator Panel Weight 61.0 lbs

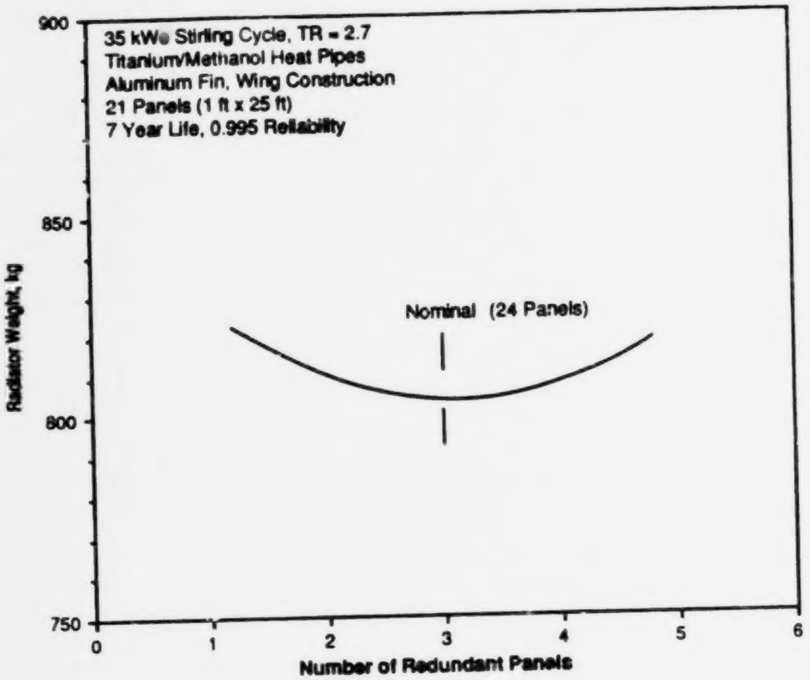


Figure 6.4.5-1. 35 kWe HP Stirling Radiator Weight - Effect of Redundancy

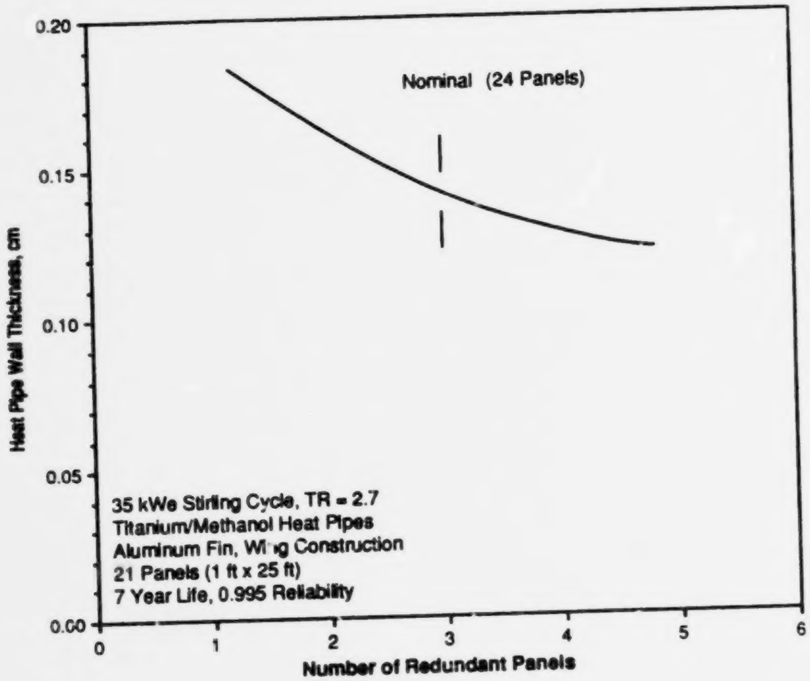


Figure 6.4.5-2. 35 kWe HP Stirling Radiator Wall Thickness Requirements

### 6.4.6 Excess Power Management Engine Operation

Engine operation was examined for the circumstance where the excess solar energy (see table 6.4.1-1) is to be processed through the engine. Both the 35 kWe mission similar to Space Station, and the 7 kWe mission which has some orbits with no eclipse, were examined for the CBC and heat pipe Stirling power system designs.

The engine efficiencies were not intentionally derated under maximum power conditions, such as with a recuperator bypass for the CBC. Rather, the increase in engine power was analyzed using the system design codes, increasing power to the point where energy consumption was increased an appropriate amount to balance consumption plus losses against the available energy input. Engine temperature ratio was decreased in each instance, causing a reduction in engine efficiency, by increase of engine outlet temperature and radiator temperature so that the calculated radiator area would duplicate the nominal design value. This accomplished the main purpose of the analysis, which was to determine the increase in radiator temperature due to increase in waste heat load between nominal design conditions and maximum solar energy conditions. Excess electric power was assumed to be dissipated by the parasitic load radiator which is included in the design as part of the engine control scheme.

Tables 6.4.6-1 and 6.4.6-2 present tabulations of selected system operating conditions at nominal design conditions and estimated for the maximum solar energy conditions. The tabulations include both the CBC and the heat pipe Stirling cycles.

Table 6.4.6-1. Excess Power Affect on CBC System Operation

| | 35 kWe | | 7 kWe | |
|---|---|---|---|---|
| | Nominal Design | Maximum Power | Nominal Design | Maximum Power |
| Excess energy ratio | 1.0 | 1.220 | 1.0 | 1.573 |
| Solar reflected to receiver, kWt | 201.9 | 216.5 | 39.9 | 42.6 |
| Solar multiple | 1.607 | 1.413 | 1.467 | 1.0 |
| Net from receiver to PCU, kWt | 110.5 | 134.8 | 24.0 | 37.7 |
| Radiator waste heat*, kWt | 74.8 | 92.2 | 16.8 | 26.7 |
| Net power, kWe | 35.0 | 41.8 | 7.00 | 10.7 |
| Turbine inlet temperature, K | 1086 | 1078 | 1086 | 1066 |
| Compressor inlet temperature, K | 300 | 319 | 285 | 321 |
| Radiator panel peak temperature, K | 375 | 397 | 357 | 398 |

*Note: Includes both PCU and electronic components cooling loads

Table 6.4.6-2. Excess Power Affect on HP Stirling System Operation

| | 35 kWe | | 7 kWe | |
|---|---|---|---|---|
| | Nominal Design | Maximum Power | Nominal Design | Maximum Power |
| Excess energy ratio | 1.0 | 1.220 | 1.0 | 1.573 |
| Solar reflected to receiver, kWt | 172.6 | 185.1 | 35.0 | 37.5 |
| Solar multiple | 1.607 | 1.413 | 1.467 | 1.0 |
| Net from receiver to PCU, kWt | 94.5 | 117.0 | 21.0 | 33.8 |
| PCU radiator waste heat, kWt | 55.4 | 72.1 | 12.9 | 22.5 |
| Net power, kWe | 35.0 | 40.4 | 7.0 | 10.2 |
| Engine hot temperature, K | 1033 | 1012 | 1033 | 1012 |
| Engine cold temperature, K | 383 | 422 | 356 | 435 |
| Engine temperature ratio | 2.70 | 2.40 | 2.90 | 2.33 |
| Radiator panel peak temperature, K | 364 | 396 | 337 | 398 |

# 7.0 SOLAR DYNAMIC POWER SYSTEMS CONCEPTUAL DESIGNS

The work reported in this section corresponds to the conceptual design activity of Task III from the SOW.

## 7.1 OBJECTIVE

Based on the results of Tasks I and II using state-of-the-art technology, develop a conceptual design (Conceptual and Design Level Drawing (Level I)) and configuration of the solar Brayton, Rankine, and Stirling power system for each mission power level category. This shall identify the size, weight, configuration, and view factors where appropriate.

## 7.2 DESIGN DESCRIPTIONS

A total of six conceptual designs were prepared, three each at 35 kWe and 7 kWe power levels. The three designs for each power level are: the CBC, the heat pipe Stirling, and the pumped loop Stirling. As discussed in section 2.3.4, the alkali-metal Rankine cycle was eliminated as a result of the Task II trade studies.

The conceptual design drawings illustrate component arrangement and layout within the system. Overall dimensions are indicated on the system layout drawings. Detailed size and weight information is presented in tables included in section 6.4.

The first six figures (7.2-1 through 7.2-6) are isometric renderings of integrated receiver/TES/PCU packages. The CBC configuration is similar to figure 7.2-1, an illustration of the Space Station CBC design taken from an earlier issue of DR-02. The 35 kWe CBC receiver/TES design for this study was a composite design comprised of the Garrett CBC shell design (ref. 1) adjusted for size and higher temperature, and the internal design of the Boeing A/D receiver (L.M. Sedgwick, Boeing, Contract NAS3-24669) adjusted from 24 to about 21 HX/TES tubes of 0.102 m (4.0 in.) diameter. The 7 kWe CBC design would be similar in layout with about 12 HX/TES tubes of 0.089 m (3.5 in.) diameter, although no detailed rendering was prepared; an outline of the 7 kWe

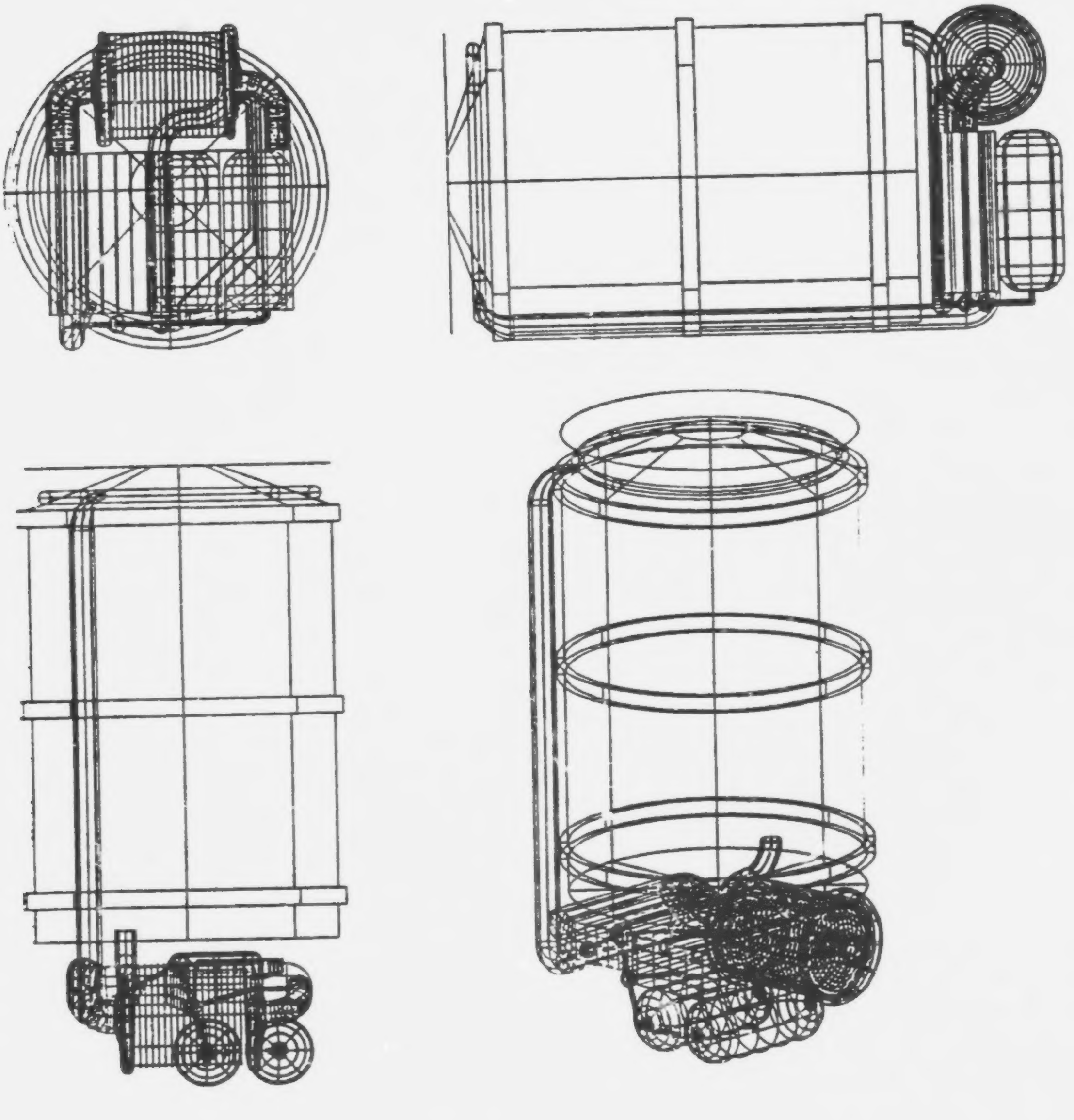


Figure 7.2-1. CBC Receiver/PCU Integrated Package

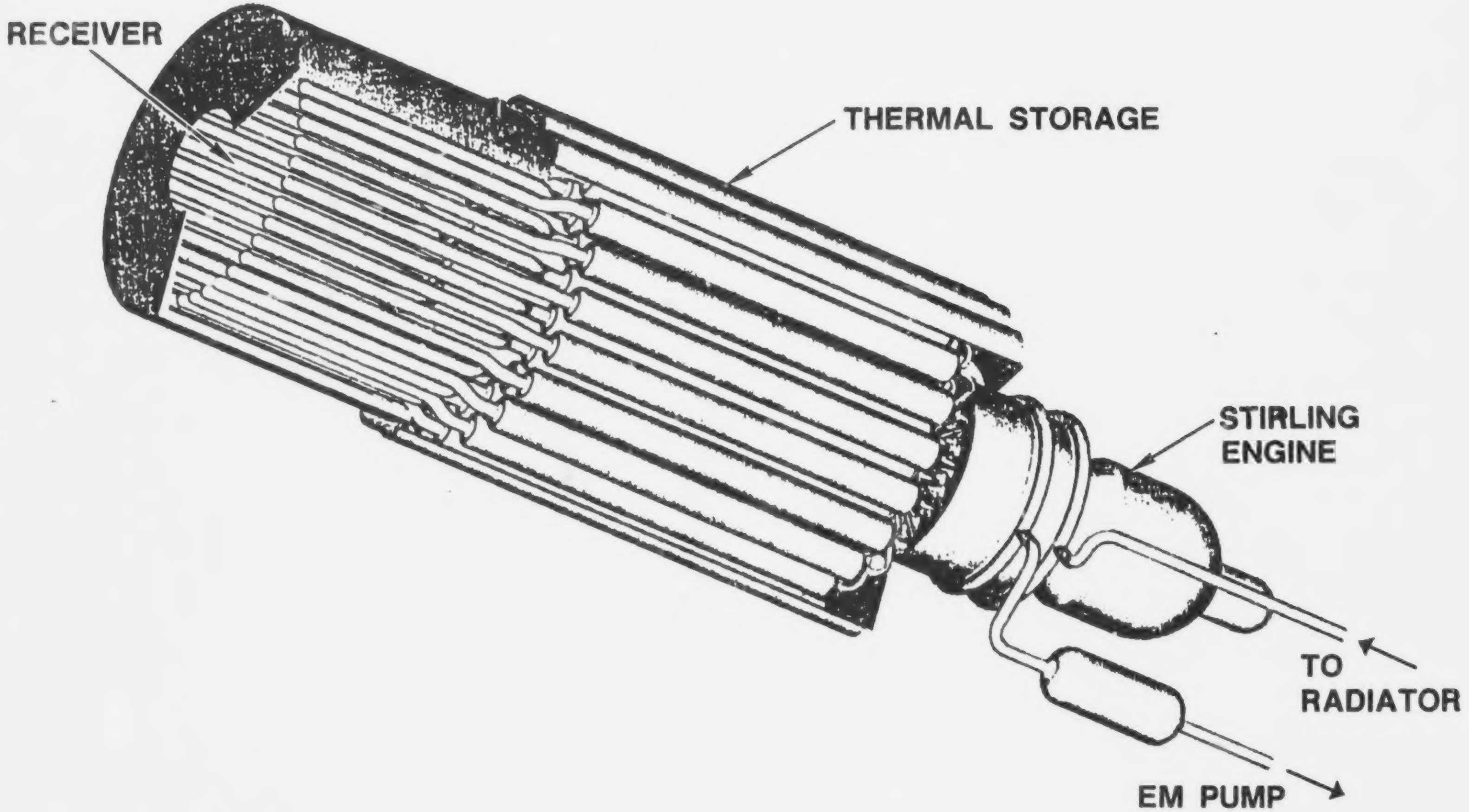

RECEIVER
THERMAL STORAGE
STIRLING ENGINE
TO RADIATOR
EM PUMP

Figure 7.2-2. 35 kWe Heat Pipe Stirling Engine/Solar Receiver

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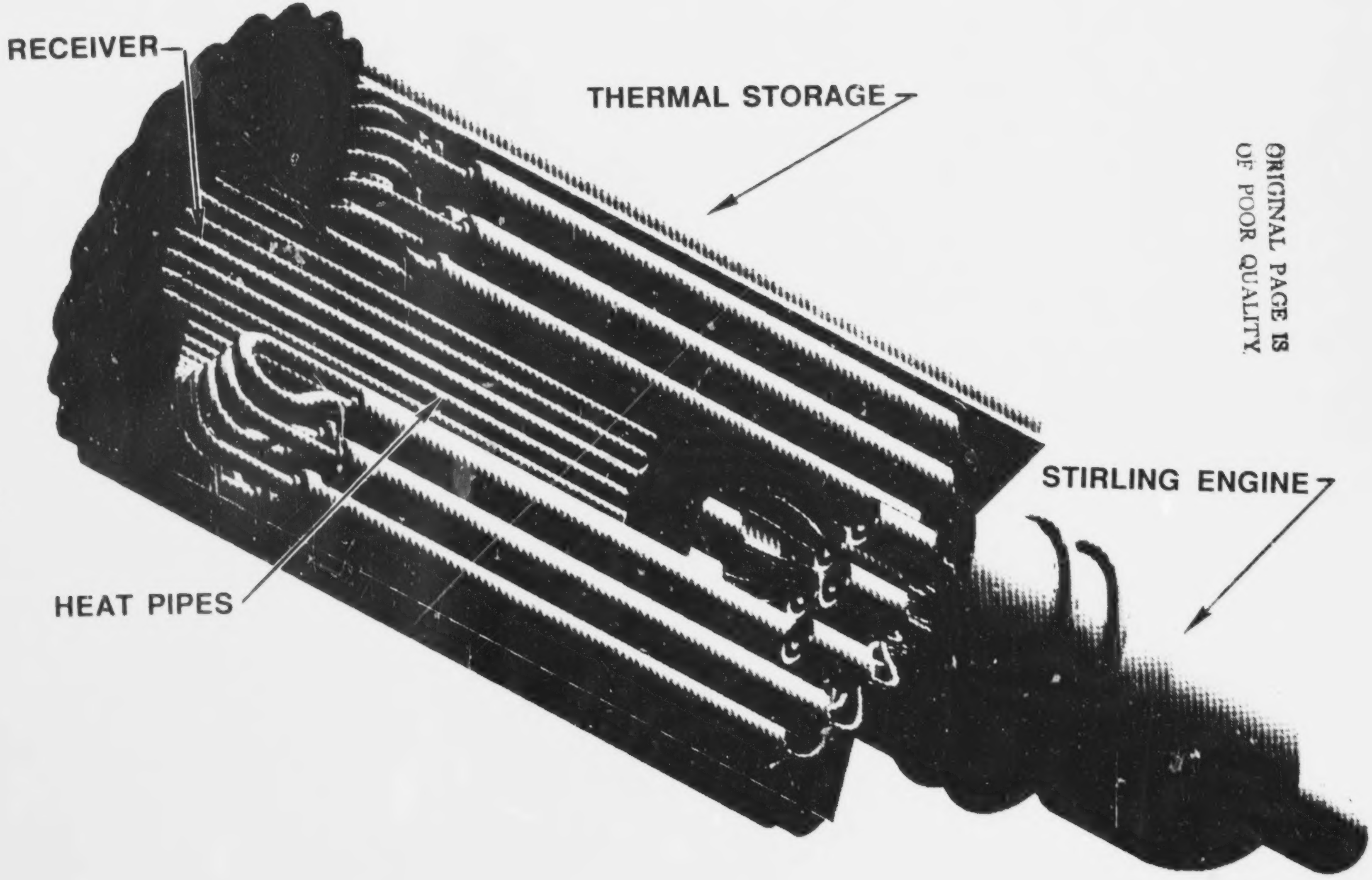
RECEIVER
THERMAL STORAGE
HEAT PIPES
STIRLING ENGINE

Figure 7.2-3. 35 kWe Heat Pipe Stirling Engine/Solar Receiver (Design Variation)

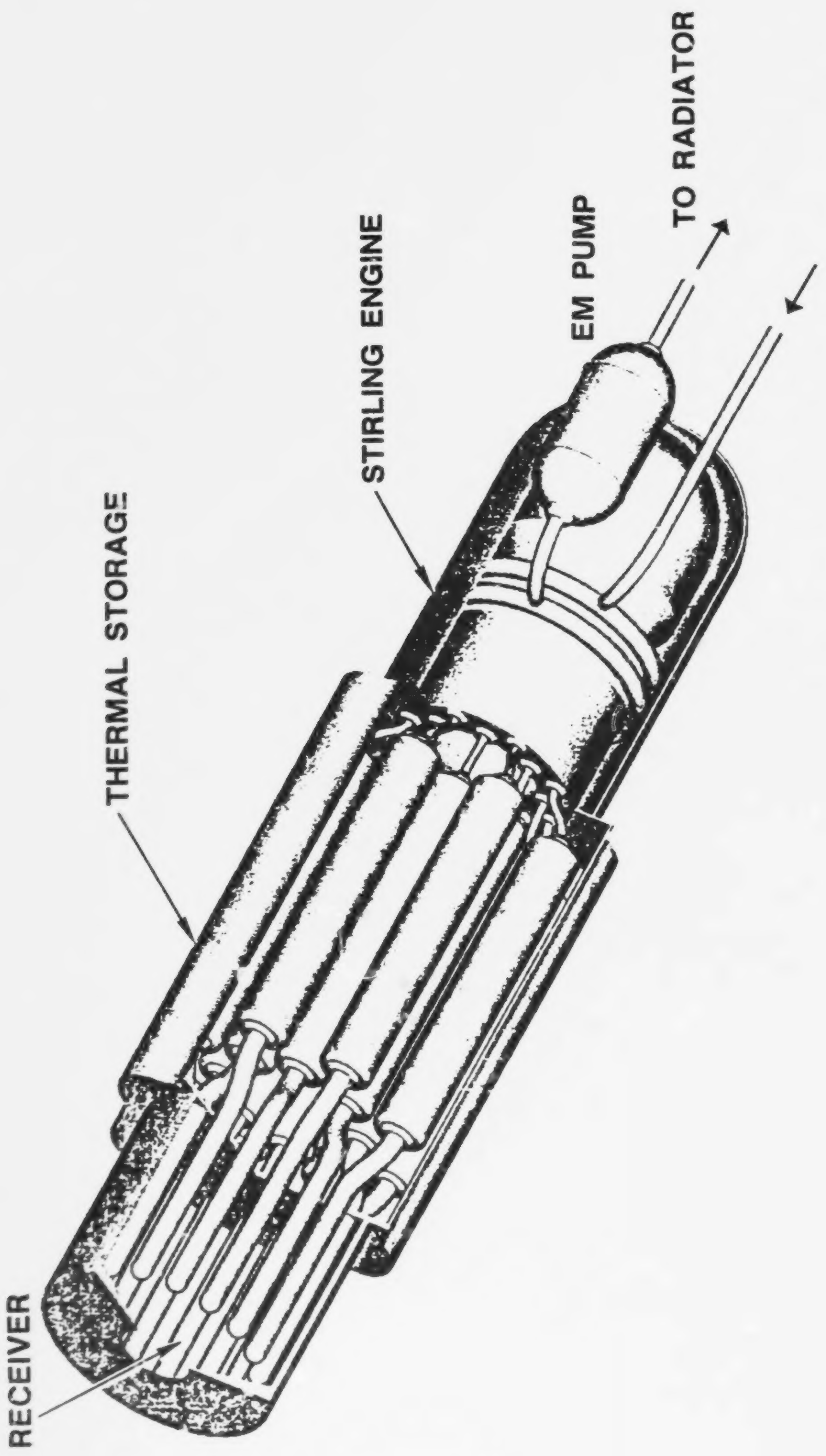
RECEIVER
THERMAL STORAGE
STIRLING ENGINE
EM PUMP
TO RADIATOR

Figure 7.2-4. 7 kWe Heat Pipe Stirling Engine/Solar Receiver

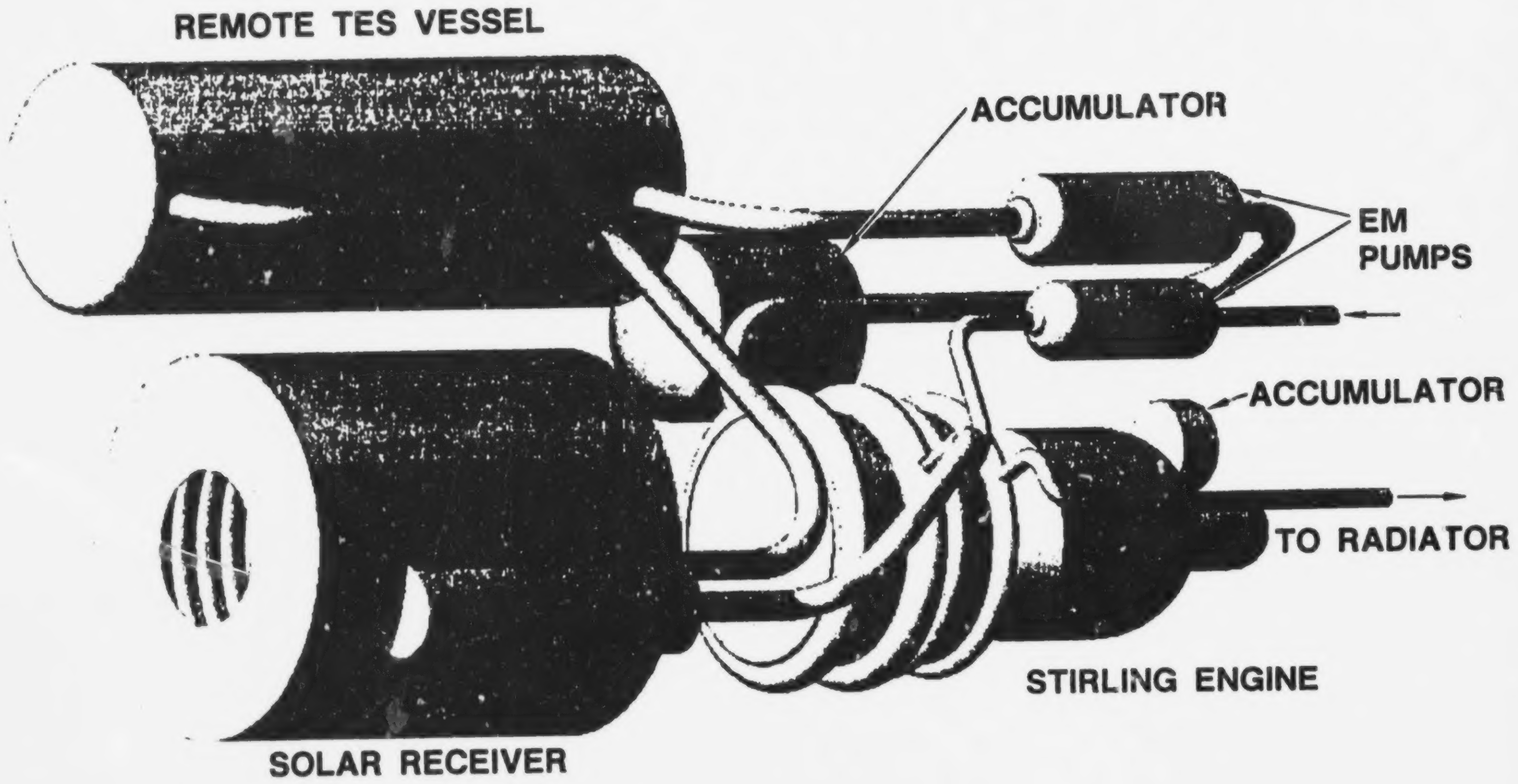
REMOTE TES VESSEL
ACCUMULATOR
EM
PUMPS
ACCUMULATOR
TO RADIATOR
STIRLING ENGINE
SOLAR RECEIVER

Figure 7.2-5. 35 kWe Stirling Engine/Pumped Loop Receiver

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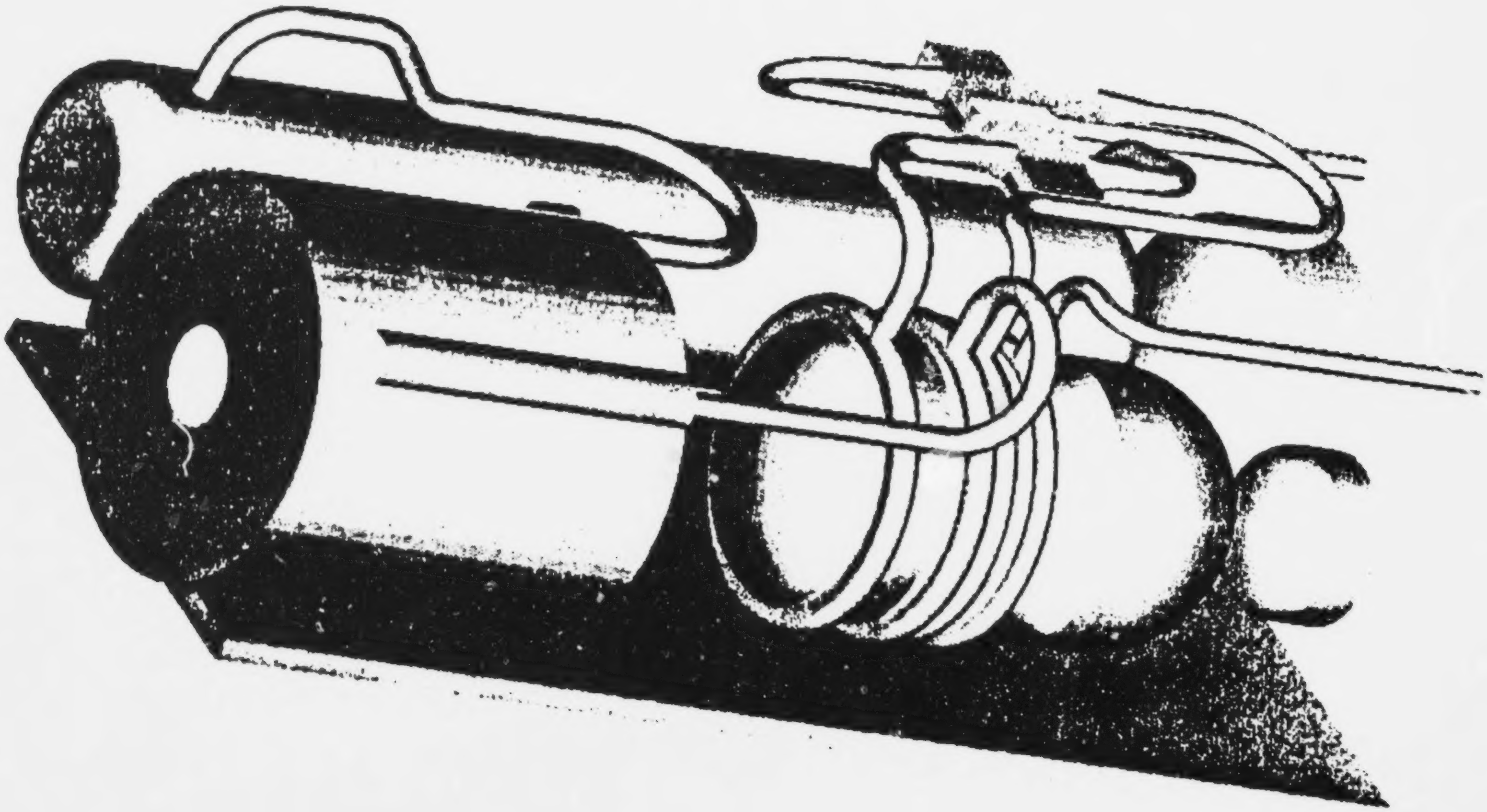

Figure 7.2-6. 7 kWe Stirling Engine/Pumped Loop Receiver

unit appears on the later system layout drawing.

Figures 7.2-2, 7.2-3, and 7.2-4 illustrate the heat pipe Stirling configurations. The 35 kWe design shown in figure 7.2-2 utilizes 40 heat pipe/TES units, each TES unit being 1.09 m long by 0.095 m in diameter (43 in. by 3.75 in.). The TES units are arranged in two concentric rows of 20 each. A design variation for the 35 kWe system is shown in figure 7.2-3, wherein the receiver heat pipes were designed with a ~180 degree bend so as to lie within the space formed by the TES units, resulting in a shorter overall assembly. The design concept looks interesting, but the design is basically just a sketch, and no thermal analysis of the design was performed.

The 7 kWe heat pipe Stirling design shown in figure 7.2-4 utilizes 16 heat pipe/TES units, each TES unit being 0.59 m long by 0.095 m in diameter (23 in. by 3.75 in.). The TES units are arranged in two concentric rows of 8 each.

The 35 kWe pumped loop Stirling design is shown in figure 7.2-5. The remote TES vessel is 1.40 m long (55 in.) and the overall assembly is 2.78 m (108 in.) long. The assembly frontal profile, including a component mounting frame, would fit within a rectangle 1.10 m by 1.60 m (43 in. by 63 in.). The 7 kWe pumped loop Stirling design is quite similar in layout. The remote TES vessel uses the same TES cannister design as for the larger engine design, so is of the same length, 1.40 m (55 in.). The envelope for this assembly is 1.68 m (66 in.) length, with a frontal profile which would fit within a rectangle 0.63 m by 0.77 m (25 in. by 30 in.). This illustration of the 7 kWe package shows an arrangement with TEM (thermoelectric electromagnetic) pumps, whereas the final design actually included ALIP EM pumps similar to the 35 kWe engine.

The next six figures (7.2-7 through 7.2-12) present several views of each of the six solar dynamic power systems. All six feature the truss hex concentrator design and heat pipe radiator design. The radiator size for each of the two CBC applications has been increased to handle the additional electronic components waste heat load. The radiator coolant return temperature is low enough for CBC to provide for the electronic cooling load. Required area for the estimated electronic heat load amounted to four panels

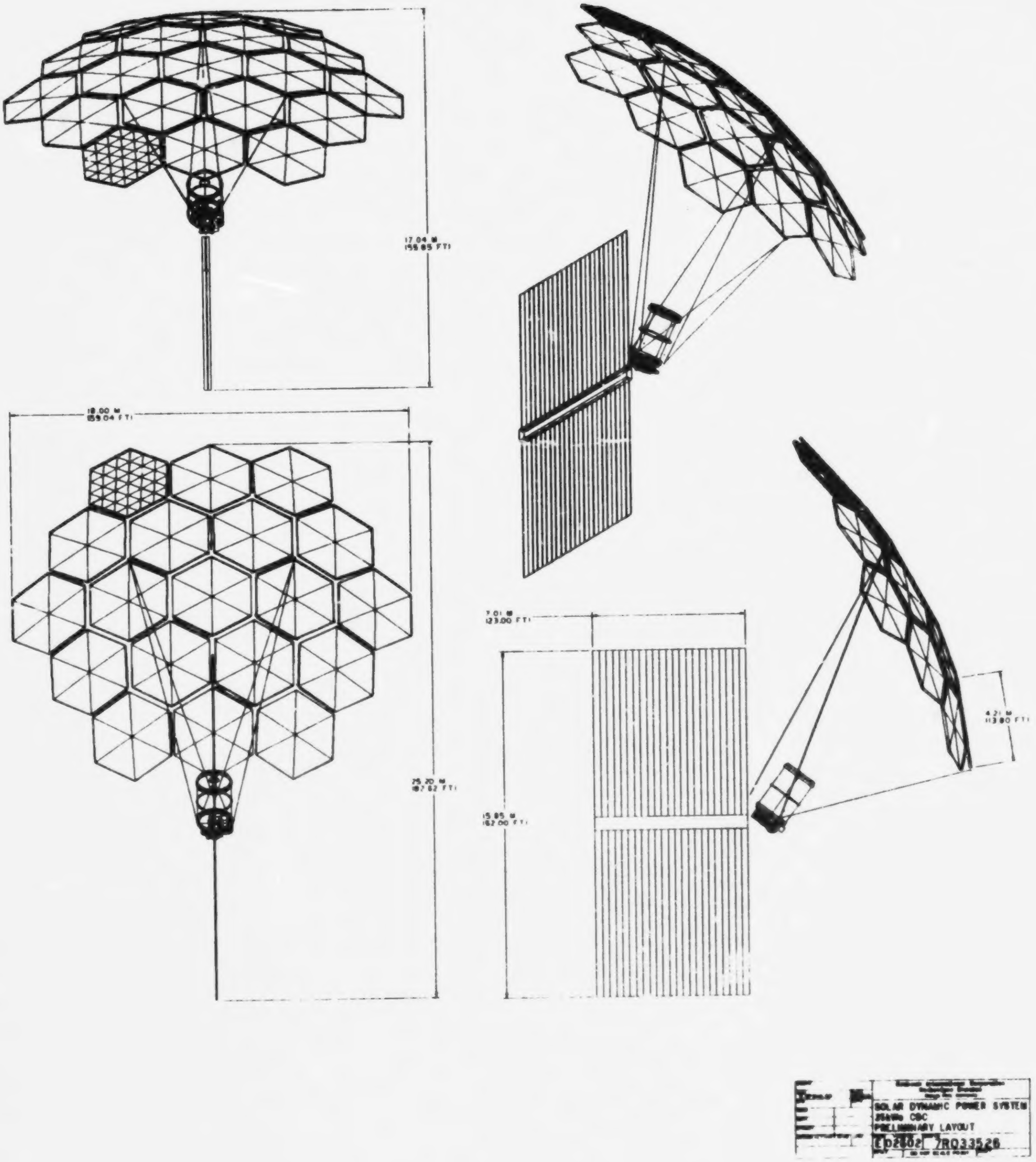
17.04 M
(55.85 FT)
18.00 M
(59.04 FT)
25.20 M
(82.62 FT)
7.01 M
(23.00 FT)
15.85 M
(52.00 FT)
4.21 M
(13.80 FT)
SOLAR DYNAMIC POWER SYSTEM
35kWe CBC
PRELIMINARY LAYOUT
E D2802 7RO33526

Figure 7.2-7. System Layout 35 kWe CBC

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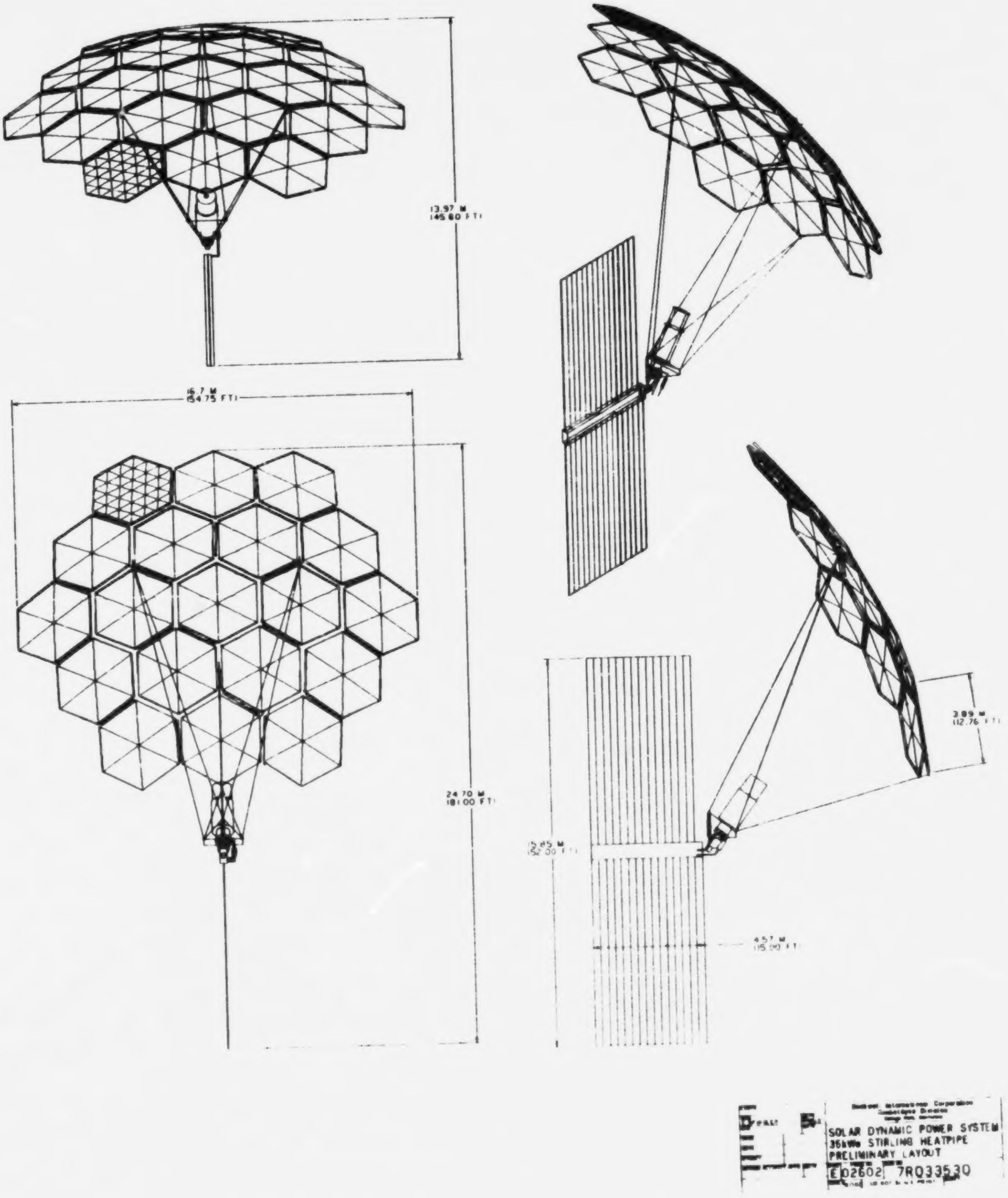
13.97 M
(45.80 FT)
16.7 M
(54.75 FT)
24.70 M
(81.00 FT)
15.85 M
(52.00 FT)
3.89 M
(12.76 FT)
4.57 M
(15.00 FT)
SOLAR DYNAMIC POWER SYSTEM
35kWe STIRLING HEATPIPE
PRELIMINARY LAYOUT
E 02602 7RO33530

Figure 7.2-8. System Layout 35 kWe Heat Pipe Stirling

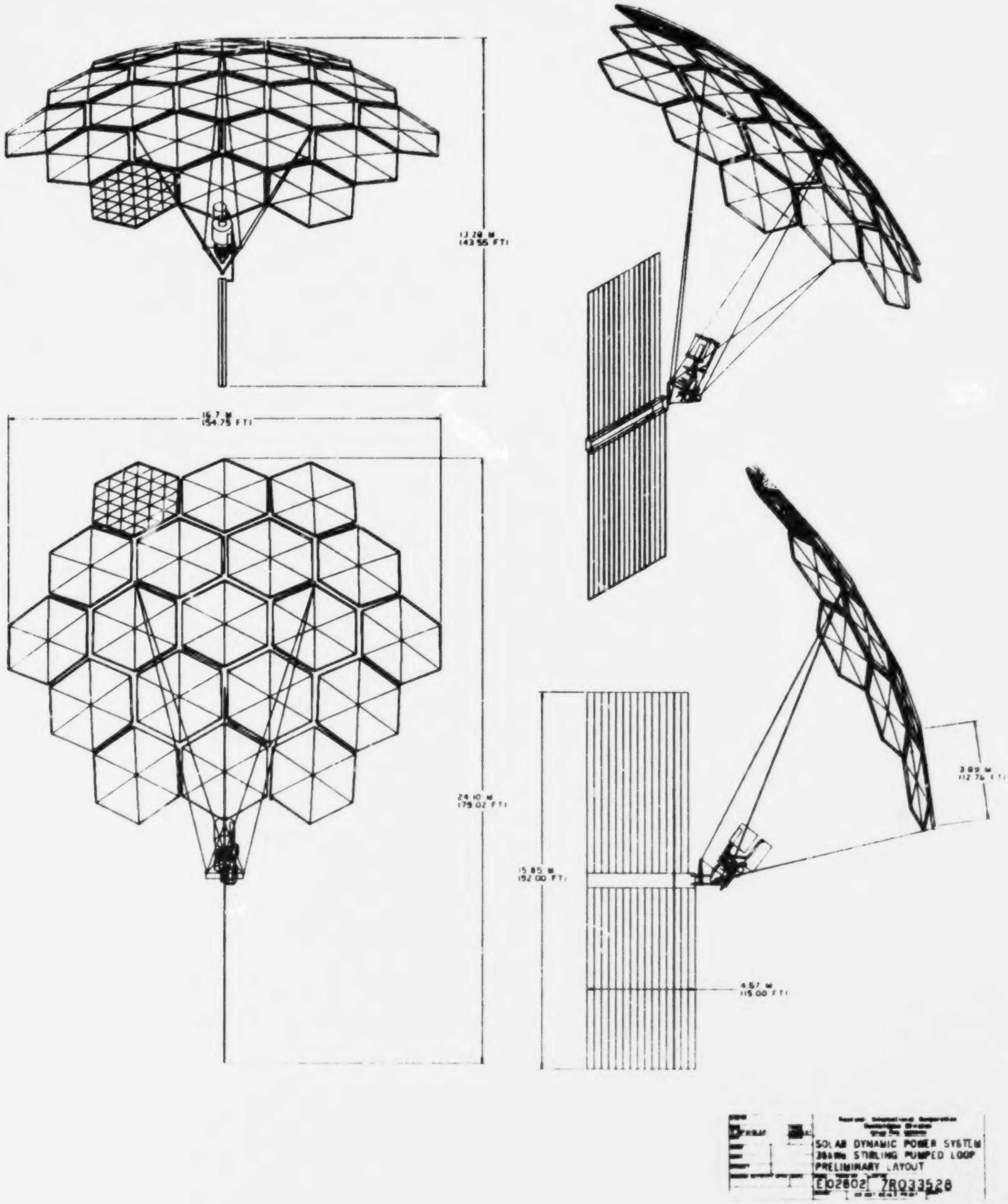
13.28 M (43.55 FT)
16.7 M (54.75 FT)
24.10 M (79.02 FT)
3.89 M (12.76 FT)
15.85 M (52.00 FT)
4.57 M (15.00 FT)
SOLAR DYNAMIC POWER SYSTEM
35kWe STIRLING PUMPED LOOP
PRELIMINARY LAYOUT
E02602 7RO33528

Figure 7.2-9. System Layout 35 kWe Pumped Loop Stirling

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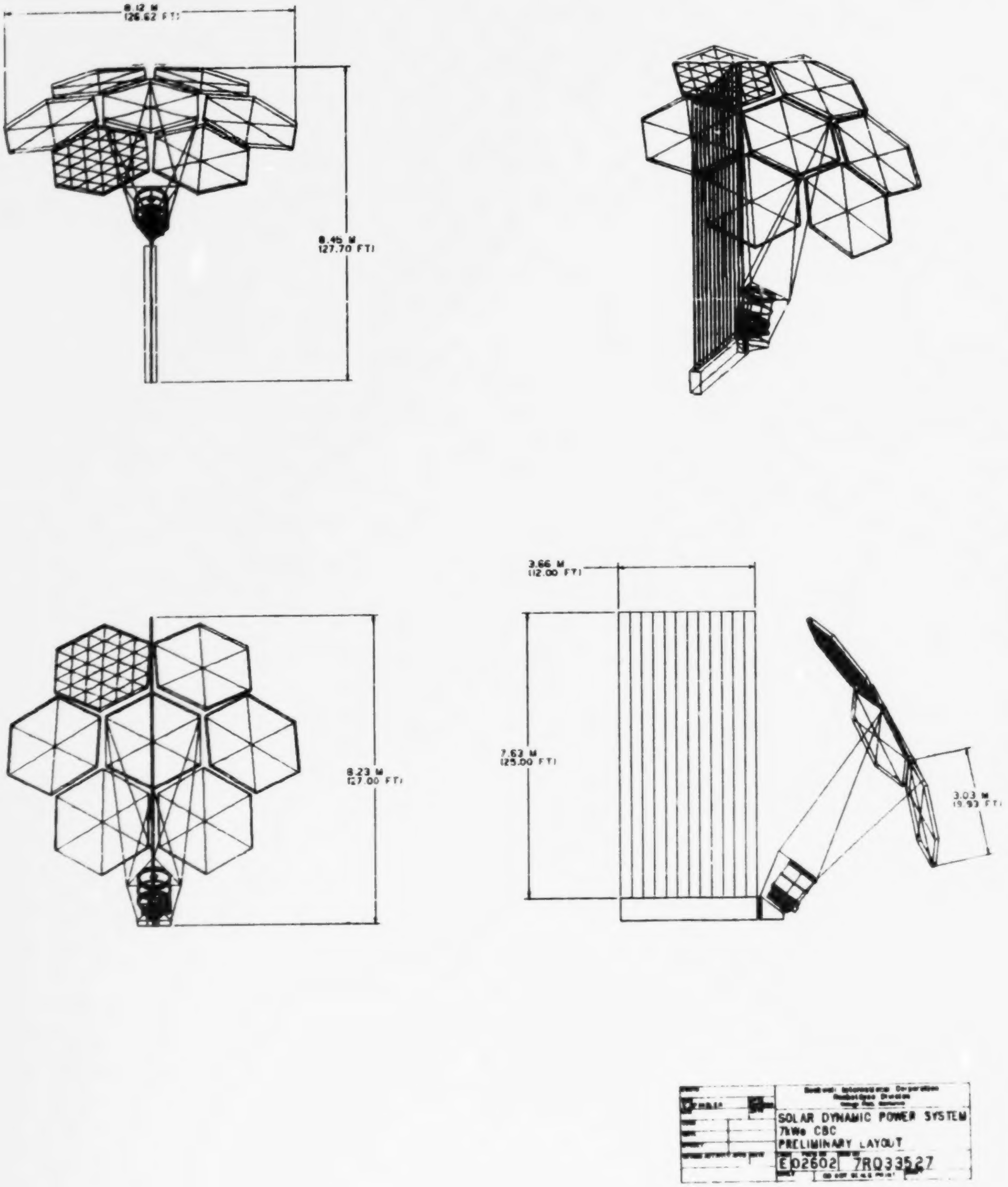
8.12 M
(26.62 FT)
8.45 M
(27.70 FT)
3.66 M
(12.00 FT)
8.23 M
(27.00 FT)
7.62 M
(25.00 FT)
3.03 M
(9.93 FT)
SOLAR DYNAMIC POWER SYSTEM
7kWe CBC
PRELIMINARY LAYOUT
E 02602
7RO33527

Figure 7.2-10. System Layout 7 kWe CBC

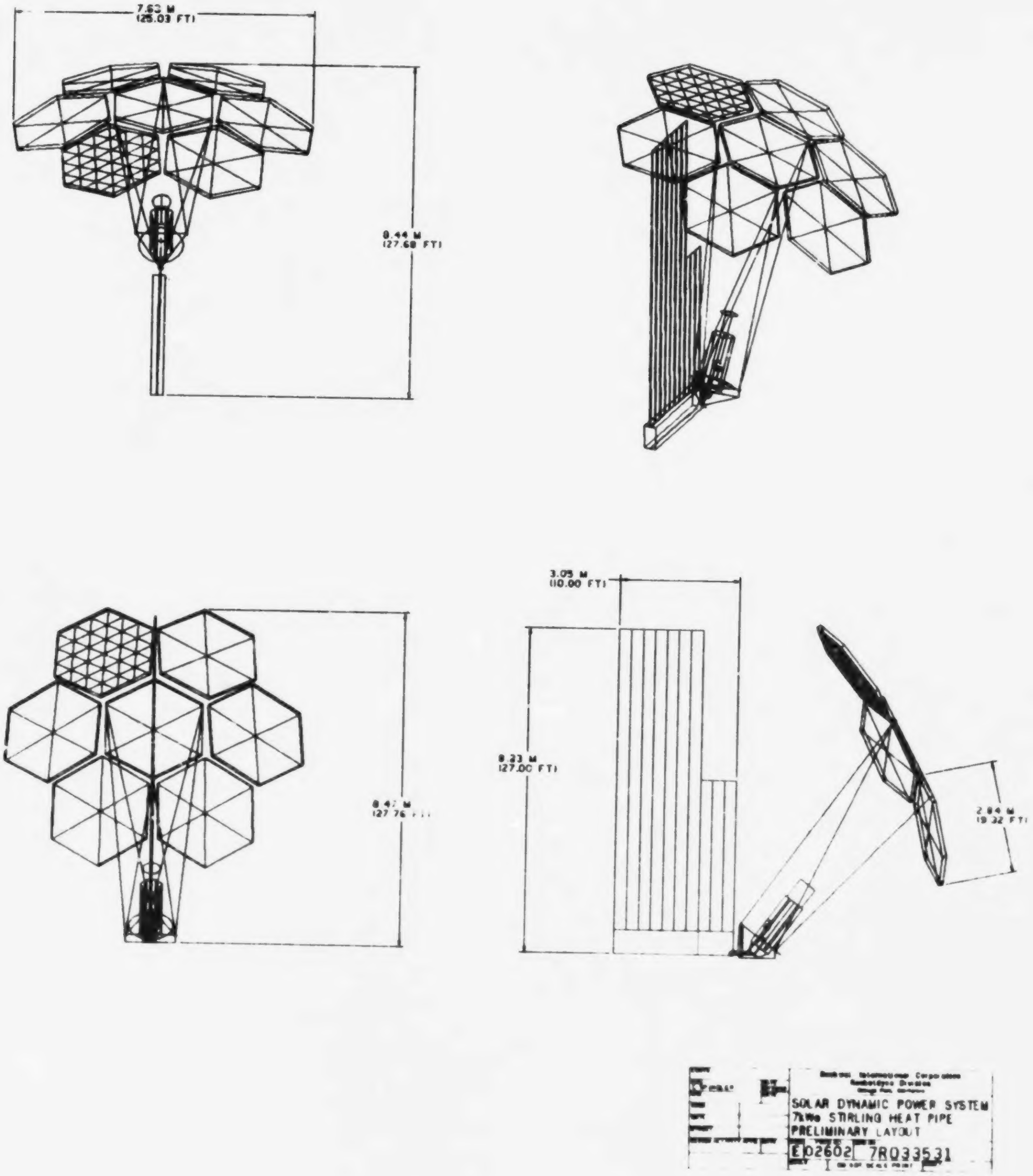
7.63 M
(25.03 FT)
8.44 M
(27.68 FT)
3.05 M
(10.00 FT)
8.23 M
(27.00 FT)
8.47 M
(27.76 FT)
2.84 M
(9.32 FT)
Rockwell International Corporation
Rocketdyne Division
SOLAR DYNAMIC POWER SYSTEM
7kWe STIRLING HEAT PIPE
PRELIMINARY LAYOUT
E 02602 7R033531

Figure 7.2-11. System Layout 7 kWe Heat Pipe Stirling

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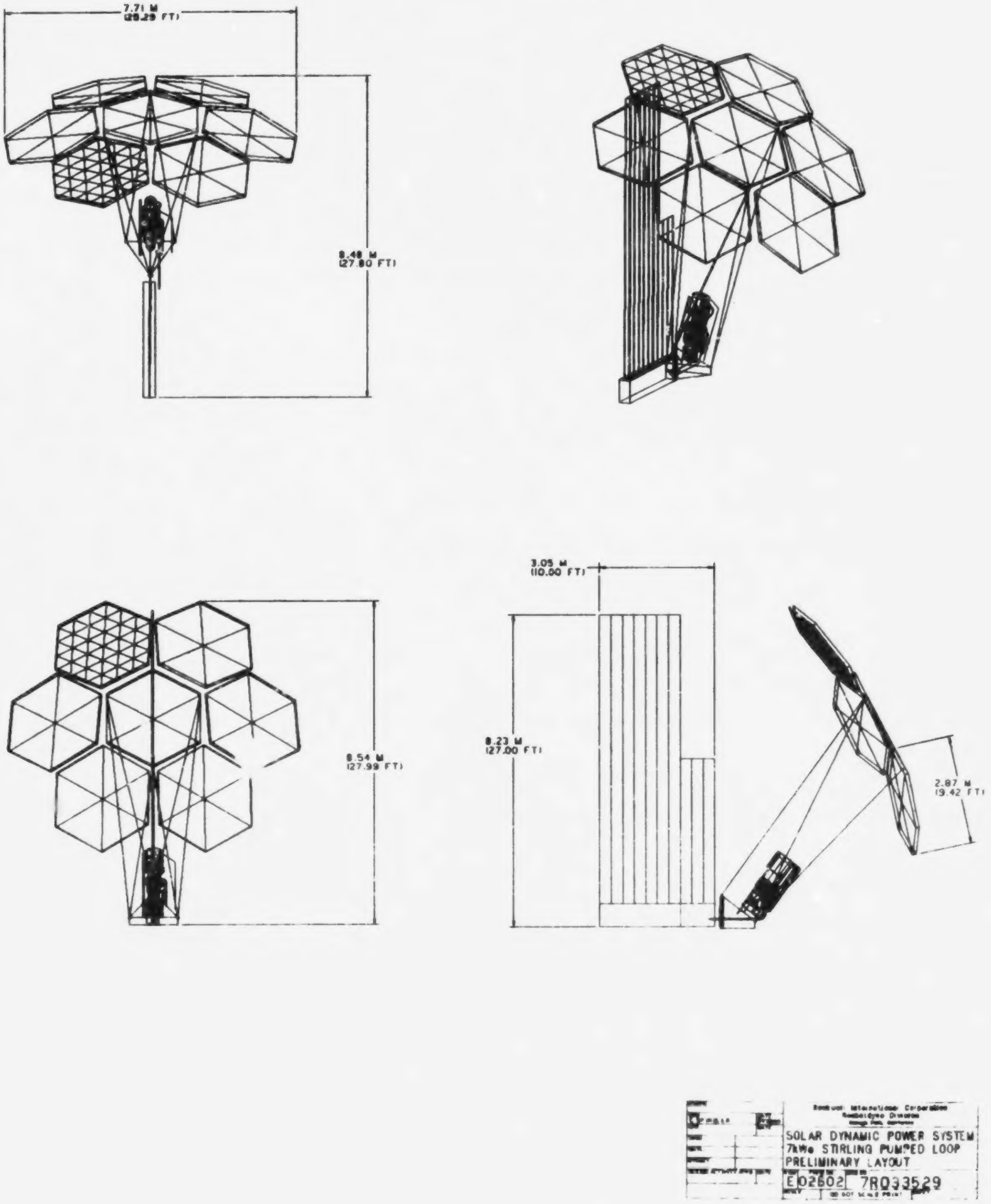

7.71 M
(25.29 FT)
8.48 M
(27.80 FT)
3.05 M
(10.00 FT)
8.23 M
(27.00 FT)
8.54 M
(27.99 FT)
2.87 M
(9.42 FT)
Rockwell International Corporation
Rocketdyne Division
SOLAR DYNAMIC POWER SYSTEM
7kWe STIRLING PUMPED LOOP
PRELIMINARY LAYOUT
E 02602
7R033529
DO NOT SCALE PRINT

Figure 7.2-12. System Layout 7 kWe Pumped Loop Stirling

at 35 kWe and 1.1 panels at 7 kWe. Redundancy requirements for CBC are figured in as part of the radiator as a whole, approximately 15% additional area for 7-year lifetime and 0.995 reliability.

For the Stirling cycle designs, the engine cooling fluid operates at temperatures which are too high for electronic component cooling purposes, so a separate radiator would be required. For the 35 kWe system, a five-panel radiator is shown adjacent to the engine waste heat radiator. For the 7 kWe systems, which only needed 1.1 full size panels for cooling purposes, a smaller three-panel design is shown using 50%-60% length panels which includes one panel for redundancy.

Other features included in the power systems but which are not apparent from the conceptual design drawings are: the parasitic load radiator, and the interface adaptor and superstructure assembly (to which all of the subsystems are attached). The designs do not include the 2-axis concentrator vernier pointing gimbal which is planned for Space Station application.

The final two figures (7.2-13 and 7.2-14) illustrate the truss hex concentrator layouts; 19 hexes for the 35 kWe system and 7 hexes for the 7 kWe system. The CBC 19-hex concentrator hex size slightly exceeds the Space Station CBC design at 4.21 m point-to-point, with the Stirling cycles at 3.89 m. The 7-hex concentrator hex size would be 3.03 m for CBC, with the Stirling cycles at 2.84 m. The number of facets per hex are shown as 54 for both the 35 kWe 19-hex design and for the 7 kWe 7-hex design in recognition of the higher concentration ratio and smaller receiver aperture diameters for these designs as compared to the Space Station designs.

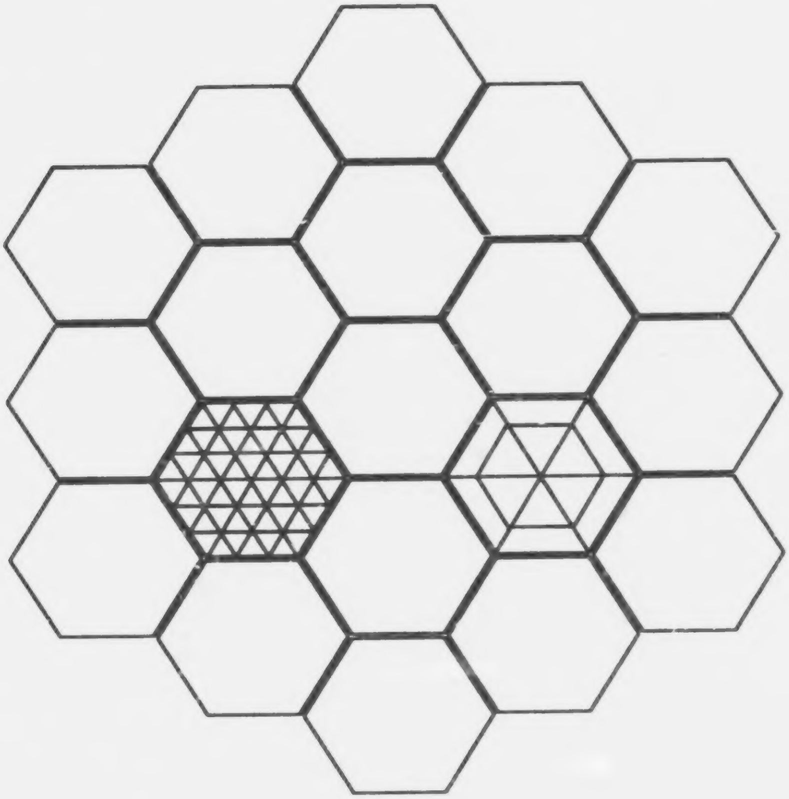


Figure 7.2-13. 35 kWe 19-Hex Concentrator

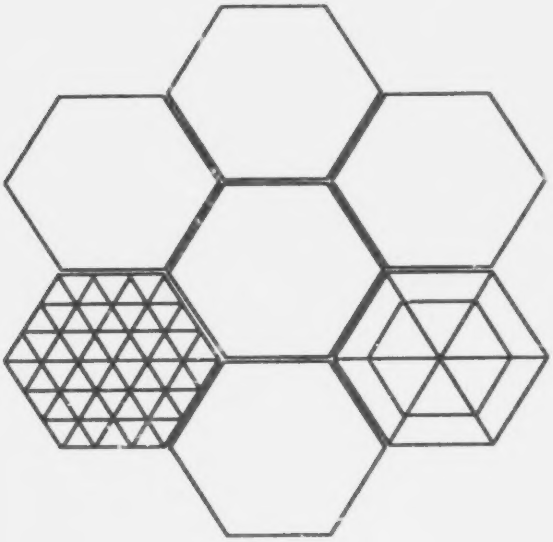


Figure 7.2-14. 7 kWe 7-Hex Concentrator

# 8.0 SOLAR DYNAMIC POWER SYSTEMS RANKING

The work reported in this section corresponds to the power system ranking activity of Task III from the SOW.

## 8.1 OBJECTIVE

The study results shall be reviewed to determine a ranking of the application potential of the Brayton, Rankine, and Stirling power systems for each mission. The criteria for ranking the power systems shall be submitted for NASA review/approval.

## 8.2 RANKING CRITERIA

The purpose of the ranking study was to compare the power system designs on other than a weight and area basis. Discriminators were identified which lead to the recommended evaluation criteria and weightings presented in table 8.2-1.

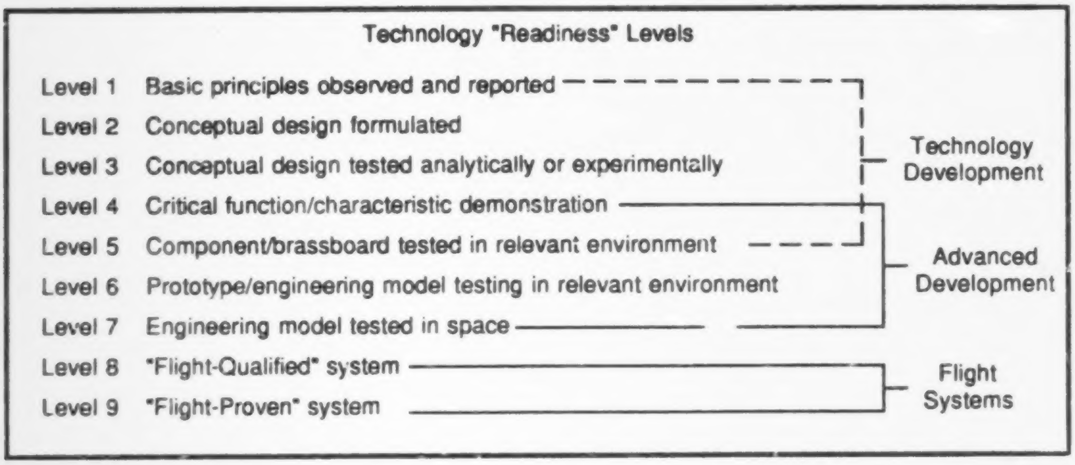
The evaluation criteria chosen were a combination of quantitative and qualitative discriminators. The relative rankings for each of the criterion were established generally as follows:

1. A number of subcriteria were developed for each criterion, as indicated in table 8.2-1.
2. Quantitative differences between cycles were determined for each subcriterion for performance and estimated life cycle cost. Technology readiness for each cycle was selected from the Technology Status Scale (table 8.2-2) which was recently developed by NASA, and which was subsequently used for the Space Station proposals.
3. Qualitative differences between the cycles were determined by comparing the two Stirling cycles to the Brayton cycle for each subcriterion; as being similar, better, worse, much better, or much worse.

## Table B.2-1. Recommended Solar Dynamic Power System Evaluation Criteria and Weights

| Unserviceable Solar Dynamic Power System (No Resupply, 1200 km Altitude) | | Serviceable Solar Dynamic Power System (With Resupply, 500 km Altitude) | |
|---|---|---|---|
| Criteria | Weighting Factor | Criteria | Weighting Factor |
| Life cycle cost<br>Development<br>Flight hardware<br>Launch | 8 | Life cycle cost<br>Development<br>Flight hardware<br>Launch<br>Operation and support | 16 |
| Operability<br>Launch packaging<br>Required pointing accuracy<br>Accommodation of power growth or shrinkage<br>Ease of deployment/assembly<br>Commonality for all missions | 14 | Operability<br>Launch packaging<br>Required pointing accuracy<br>Accommodation of power growth<br>Ease of orbital assembly<br>Ease of on-orbit repair/servicing<br>Commonality for all missions | 12 |
| Performance<br>System weight<br>Micrometerorite exposure area | 14 | Performance<br>System weight<br>Drag producing area | 12 |
| Reliability/Safety<br>Redundancy requirements<br>Simplicity of design and installation<br>Resistance to degradation<br>Presence of life-limiting components<br>Fluid contamination potential<br>Micrometerorite penetration/fluid protection potential | 50 | Reliability/Safety<br>Redundancy requirements<br>Simplicity of design and installation<br>Resistance to degradation<br>Presence of life-limiting components<br>Fluid contamination potential<br>Ease of restoration after failures<br>Number and severity of safety hazards | 40 |
| Technology Readiness<br>Technology status<br>Technical uncertainty<br>Influence of uncertainty on total system | 14 | Technology Readiness<br>Technology status<br>Technical uncertainty<br>Influence of uncertainty on total system | 12 |
| | | Compatiblity<br>Compatibility with OTV, OMV, Shuttle, and/or platforms<br>Ease of conversion from initial condition to growth<br>Interaction with other system elements | 8 |
| Total | 100 | Total | 100 |

Table 8.2-2. Technology Status Scale



Technology "Readiness" Levels

| | | |
|---|---|---|
| Level 1 | Basic principles observed and reported | Technology Development |
| Level 2 | Conceptual design formulated | |
| Level 3 | Conceptual design tested analytically or experimentally | |
| Level 4 | Critical function/characteristic demonstration | Advanced Development |
| Level 5 | Component/brassboard tested in relevant environment | |
| Level 6 | Prototype/engineering model testing in relevant environment | |
| Level 7 | Engineering model tested in space | |
| Level 8 | "Flight-Qualified" system | Flight Systems |
| Level 9 | "Flight-Proven" system | |

4. A nonlinear scaling was then established to reduce the comparative results from steps (2) and (3) above to numeric ranking values for each of the subcriterion.
5. The unweighted ranking for each of the criterion was the average of the associated subcriteria rankings.
6. Multiplication of the unweighted rankings by the ranking weights produced the results presented in section 8.3.

The evaluation criteria weighting factors were established using criteria comparison sheets such as the example shown in figure 8.2-1. The survey sheets were completed by four individuals at NASA-LeRC and five individuals at Rocketdyne for both the conditions of unserviceable and serviceable spacecraft. Results of the survey are shown in figures 8.2-2 and 8.2-3.

|  | ABSOLUTELY MORE IMPORTANT ↑ |  |  |  |  |  | SOMEWHAT MORE IMPORTANT ↑ |  | = ↑ |  | SOMEWHAT MORE IMPORTANT ↑ |  |  |  |  |  | ABSOLUTELY MORE IMPORTANT ↑ |  |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
|  | 9 | 8 | 7 | 6 | 5 | 4 | 3 | 2 | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 |  |
| LIFE CYCLE COSTS |  |  |  | X |  |  |  |  |  |  |  |  |  |  |  |  |  | OPERABILITY |
| LIFE CYCLE COSTS |  |  |  |  | X |  |  |  |  |  |  |  |  |  |  |  |  | PERFORMANCE |
| LIFE CYCLE COSTS |  |  |  |  |  |  |  | X |  |  |  |  |  |  |  |  |  | RELIABILITY/SAFETY |
| LIFE CYCLE COSTS |  |  |  |  |  |  | X |  |  |  |  |  |  |  |  |  |  | TECHNICAL READINESS |
| LIFE CYCLE COSTS |  |  |  |  |  | X |  |  |  |  |  |  |  |  |  |  |  | COMPATIBILITY |
| OPERABILITY |  |  |  |  |  |  |  |  |  |  |  |  | X |  |  |  |  | PERFORMANCE |
| OPERABILITY |  |  |  |  |  |  |  |  |  |  |  |  |  | X |  |  |  | RELIABILITY/SAFETY |
| OPERABILITY |  |  |  |  |  |  |  |  |  |  |  | X |  |  |  |  |  | TECHNICAL READINESS |
| OPERABILITY |  |  |  |  |  |  |  |  | X |  |  |  |  |  |  |  |  | COMPATIBILITY |
| PERFORMANCE |  |  |  |  |  |  |  |  |  |  |  |  | X |  |  |  |  | RELIABILITY/SAFETY |
| PERFORMANCE |  |  |  |  |  |  |  |  | X |  |  |  |  |  |  |  |  | TECHNICAL READINESS |
| PERFORMANCE |  |  |  |  |  | X |  |  |  |  |  |  |  |  |  |  |  | COMPATIBILITY |
| RELIABILITY/SAFETY |  |  |  |  |  | X |  |  |  |  |  |  |  |  |  |  |  | TECHNICAL READINESS |
| RELIABILITY/SAFETY |  |  |  | X |  |  |  |  |  |  |  |  |  |  |  |  |  | COMPATIBILITY |
| TECHNICAL READINESS |  |  |  |  |  |  | X |  |  |  |  |  |  |  |  |  |  | COMPATIBILITY |

SERVICEABLE SD POWER SYSTEM
(WITH RESUPPLY)

Figure 8.2-1. Criteria Comparison

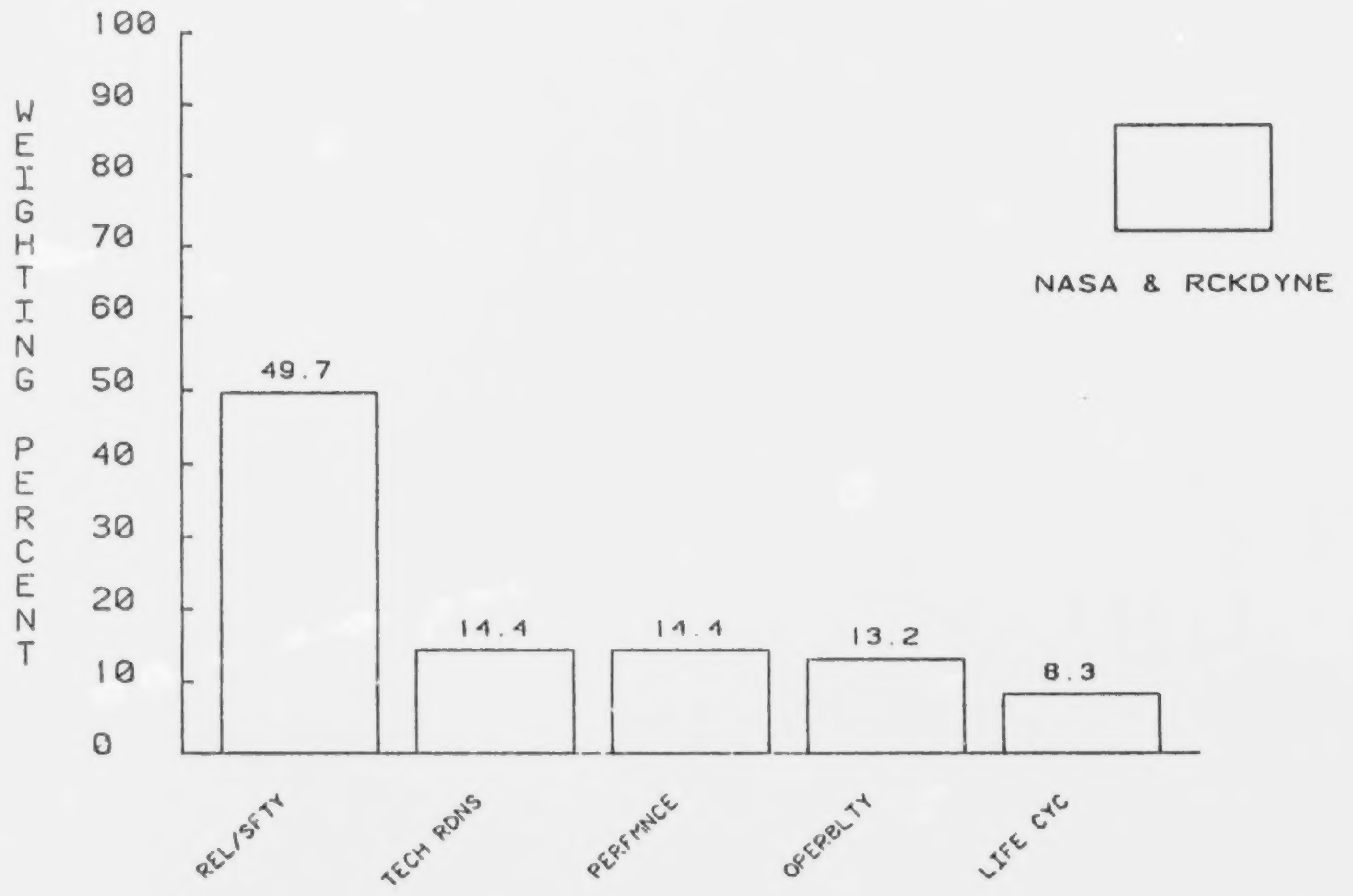


Figure 8.2-2. Criteria Weighting (without resupply)

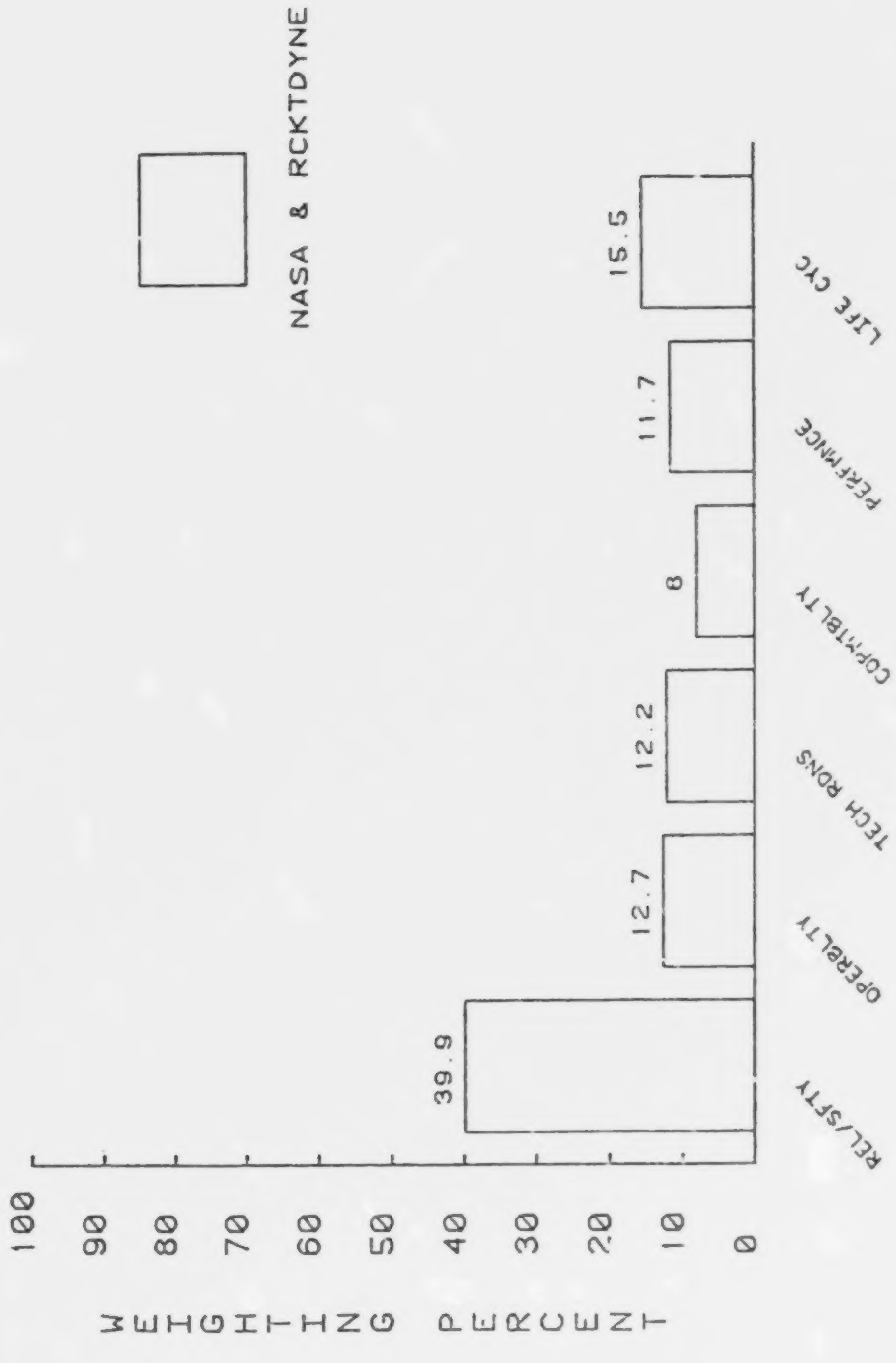


Figure 8.2-3. Criteria Weighting (with resupply)

## 8.3 TECHNICAL ASSESSMENT AND RANKING

Results of the ranking activity are presented in table 8.3-1. The bottom line ranks the heat pipe receiver Stirling cycle highest, followed by the pumped loop receiver Stirling cycle and, lastly, the Brayton cycle. The bottom line numbers should not be used to indicate the degree of superiority, but simply to indicate order of ranking based on the study groundrules. The Brayton cycle design is based on a great deal of maturity as evidenced by the extensive Space Station Phase B effort. This is not true for the Stirling cycle. As that design matures, weight growth will no doubt occur such that the substantial performance ranking advantage shown in table 8.3-1 for Stirling will tend to reduce. Table 8.3-1 does, however, point out the potential power system improvements which can be attributed to the Stirling cycle.

The ranking results were dependent on subjective comparisons, nonlinear scaling factors, assumed mission application and a variety of other imprecise inputs. Change of these inputs (i.e., qualitative rankings, design refinement effecting weight, etc.) would alter the numerical rankings and possibly even the ranking order. The ranking task was not pursued further, as the results confirmed that the performance advantages of the Stirling cycles certainly warrant further research and development work for the Stirling cycle.

Table 8.3-1. Solar Dynamic Power System Ranking

| CRITERIA | 35 kWe (WITH RESUPPLY) | | | | 7 kWe (NO RESUPPLY) | | | |
|---|---|---|---|---|---|---|---|---|
| | WEIGHT | BRAYTON | P/L REC. STIRLING | H/P REC. STIRLING | WEIGHT | BRAYTON | P/L REC. STIRLING | H/P REC. STIRLING |
| RELIABILITY/SAFETY | 40 | 0 | -5.0 | 5.0 | 50 | 0 | -6.2 | 6.2 |
| TECHNOLOGY READINESS | 12 | 1.9 | -1.2 | -1.2 | 14 | 2.2 | -0.8 | -1.4 |
| PERFORMANCE | 12 | 0 | 11.4 | 12.0 | 14 | 0 | 9.8 | 14.0 |
| OPERABILITY | 12 | 0 | -0.7 | -0.5 | 14 | 0 | 0.1 | -0.3 |
| LIFE CYCLE COST | 16 | 0 | 3.5 | 2.7 | 8 | 0 | 0.5 | 1.0 |
| COMPATIBILITY | 8 | 0 | 0 | 0 | -- | -- | -- | -- |
| TOTAL | 100 | 1.9 | 8.0 | 18.0 | 100 | 2.2 | 3.4 | 19.5 |

# 9.0 PHOTOVOLTAIC POWER SYSTEMS

The work reported in this section corresponds to the analytical study of the photovoltaic power systems, which was a part of Task II from the SOW.

## 9.1 OBJECTIVE

A comparison shall be made between the dynamic power systems in this study and photovoltaic power systems for each selected power level. The comparison shall be with both SOA silicon cells (14.5% efficiency) and with advanced gallium arsenide cells with concentrators (22% efficiency). The comparison shall be based on system efficiency, size, weight, reliability, etc. Advantages and disadvantages of solar dynamic power versus photovoltaic systems shall be identified.

## 9.2 TECHNOLOGY BASE

A complete large photovoltaic power system such as has been designed for Phase I Space Station requires four major subsystems:

1. Array assembly
2. Energy storage
3. Thermal control
4. Power management and distribution (PMAD)

Each of the subsystems for this study were patterned closely after the work done for the Phase I Space Station design; in particular, the power system Phase C/D proposal (Space Station Electric Power System Design, Development, and Production (Work Package 4), Rocketdyne, RI/RD87-201P-2, 28 July 1987). The differences between the Phase B work (ref. 1) and Phase C/D were more pronounced for the photovoltaic system than for the solar dynamic systems, particularly for battery and PMAD and to a lesser degree for the array assembly. The differences increased battery weight and reduced array and PMAD weights, with little change in overall system weight.

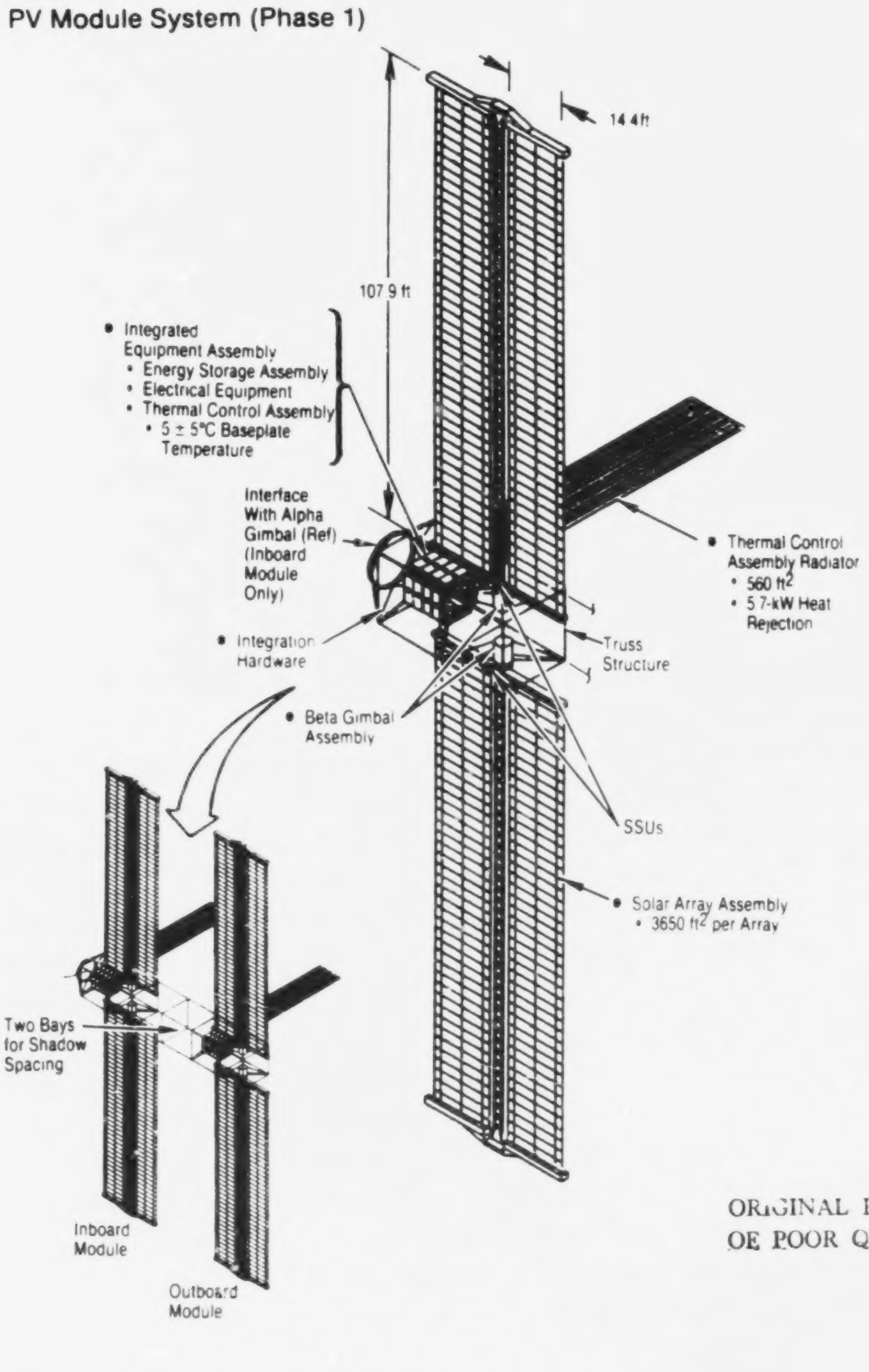
The photovoltaic power systems for this study are reasonably close to the Space Station designs:

1. The 35 kWe design is a 4-wing array about 68% of the area of one-half of the Phase I Space Station design by Lockheed Missiles and Space Company, Inc. (2 modules of 2 wings each), which is shown in figure 9.2-1.
2. The 7 kWe design is a single-wing design similar to the Space Station Polar-Orbit Platform (POP) array by Lockheed, with the single-wing array about 76% of the area of the POP, which is shown in figure 9.2-2.

The Space Station design data were used to scale the subsystem sizes and weights to the study conditions of 35 kWe and 7 kWe, 7-year life, appropriate orbital conditions, etc.

### 9.2.1 35 kWe Photovoltaic Power System

The 35 kWe power system varies little from the Space Station design except for size, a higher cell efficiency (14.5% versus 12.9% beginning-of-life (BOL)), and a 10 kg weight reduction for each orbital replacement unit (ORU). The Space Station has complete redundancy of PMAD elements for each pair of wings (each module), which for a four-wing array results in a total of four each of the several PMAD elements (dc switch unit (DCSU), main inverter unit (MIU), and photovoltaic controller (PVC)). The Space Station main bus switch unit (MBSU) and power distribution and control unit (PDCU), with a total of two each for each 4-wing array, were deleted for this study as those elements have been sized for growth of the Space Station to 175 kWe total power. The PDCU provides auxiliary power outboard of the alpha joint for operation of the beta joints, etc., and this function could be provided for an all-photovoltaic power system for less weight than planned for Space Station. The MBSU provides for isolation of the power system, whether for fault isolation or for service and repair, and that particular function has been deleted for this study as being an item related to integration of the power system with a host spacecraft.


PV Module System (Phase 1)
14.4 ft
107.9 ft
• Integrated Equipment Assembly
• Energy Storage Assembly
• Electrical Equipment
• Thermal Control Assembly
• 5 ± 5°C Baseplate Temperature
Interface With Alpha Gimbal (Ref) (Inboard Module Only)
• Thermal Control Assembly Radiator
• 560 ft²
• 5.7-kW Heat Rejection
• Integration Hardware
Truss Structure
• Beta Gimbal Assembly
SSUs
• Solar Array Assembly
• 3650 ft² per Array
Two Bays for Shadow Spacing
Inboard Module
Outboard Module

ORIGINAL PAGE IS
OF POOR QUALITY

Figure 9.2-1. Space Station Photovoltaic Power System Module

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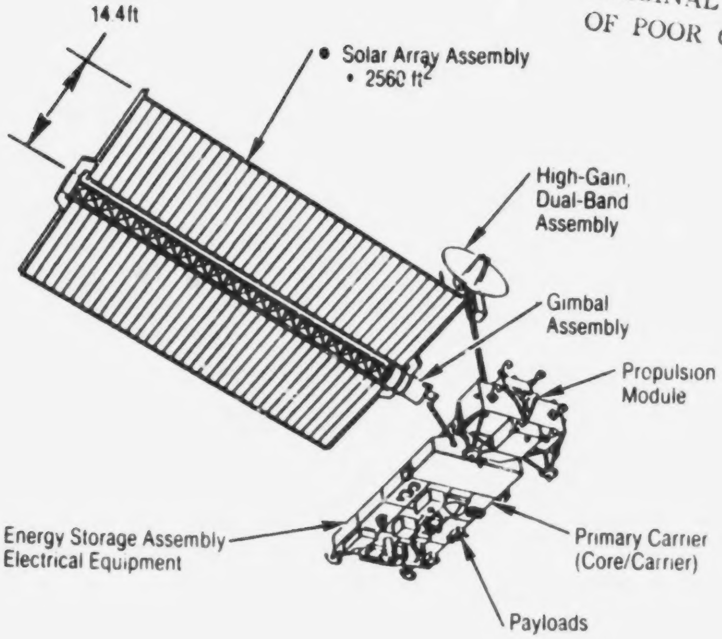

14.4ft
• Solar Array Assembly
• 2560 ft²
High-Gain, Dual-Band Assembly
Gimbal Assembly
Propulsion Module
Energy Storage Assembly Electrical Equipment
Primary Carrier (Core/Carrier)
Payloads

**Polar Orbiting (Phase 1)**

Figure 9.2-2. Space Station Polar-Orbit Platform

Thirty-cell $NiH_2$ batteries designed for a minimum capacity of 81 Ahr per battery are arranged as sets of three in series resulting in 112.5 V dc design output. Each battery set is controlled by a battery charge/discharge unit (BCDU); eight battery sets and BCDUs are required for the 35 kWe design (operating to a depth of discharge of 33.1%) versus a total of ten for half of Space Station. No battery redundancy is provided beyond exceeding the design nominal depth of discharge of $\leq$35% (for Space Station); one battery out of operation would result in 37.8% depth of discharge.

The PMAD elements and batteries for Space Station are arranged as uniform-sized packages referred to as orbital replacement units (ORU). The ORU package provides standard interface connections for power, thermal control, data and control, etc. The Space Station design allocates 27 kg (60 lb) as a standard ORU package weight for each ORU, which is added to each PMAD element and each battery. The 35 kWe and 7 kWe power systems for this study would be destined for spacecraft other than Space Station, wherein the emphasis for standardization and replaceability could well be less than for Space Station. In fact, the manner in which the Space Station platform PMAD elements will be packaged as ORUs is still uncertain. For this study, an arbitrary weight reduction of 10 kg for each ORU package was assumed to be achievable should a greater premium be attached to weight savings than was the case for the design of the Space Station. For the 35 kWe design, with 44 ORUs, the resultant weight reduction was 440 kg (970 lb) for 4 DCSUs, 4 MIUs, 4 PVCs, 8 BCDUs, and 8 battery sets (24 batteries).

The thermal control subsystem design for Space Station is a redundant pumped loop ammonia cooling assembly utilizing heat pipe radiator panels, with one assembly required for each two-wing module (see figure 9.2-1). The thermal control subsystem is designed to maintain a nominal temperature of 5±5C (278±5K) at the battery cell and electrical equipment baseplates. A major PMAD thermal input is the main inverter input. The thermal control subsystem scaling was adjusted for the lower inverter efficiencies assumed for this study (91% versus 94.5% for Space Station). Recent Space Station work has indicated that the earlier expected inversion efficiencies will not be realized (R.L. Phillips, January 1988, Rocketdyne, Canoga Park, CA, private communication).

### 9.2.2 7 kWe Photovoltaic Power System

The 7 kWe power system differs from the POP in that the POP design is presently incomplete. The PMAD elements are not packaged as ORUs such as they will be for Space Station, and no active thermal control assembly is presently planned. The 7 kWe design extends the POP designs as follows:

1. The PMAD elements (triple redundant) were assumed to be packaged as three ORUs.
2. Each power source control unit (PSCU), which includes a single battery, was assumed to be packaged as a separate ORU; a total of six are required, including one for redundancy. Use of five out of six batteries would result in depth of discharge of 32.5%.
3. An active redundant thermal control assembly scaled down from Space Station has been included in the design, whereas the current plan for the Space Station platforms would be to use passive cooling means if possible.

4. Conversion of power output to 20 kHz at 208 V ac instead of 28 V dc for the platforms.
5. The 7 kWe power system is assumed to be an unserviceable spacecraft, whereas POP is to be serviceable.

The 7 kWe design retained the POP triple redundancy for the PMAD elements except the batteries; namely, the dc control unit (DCCU), power distribution and control assembly (PDCA), and power management controller (PMC). Battery sizing resulted in five total to achieve ≤35% depth of discharge. The battery is included in the power source control unit (PSCU) along with the BCDU and main inverter. Six PSCUs are included in the 7 kWe design, allowing one for redundancy. The triple redundant DCCU, PDCA and PMC were assumed to be packaged as three separate ORU packages, with a standard package weight of 17 kg (38 lb) as was assumed for each of the 35 kWe system ORUs. The PSCU were each assumed to be packaged separately with a standard package weight of 27 kg (60 lb) for each due to the number of items in the unit, including the battery. The thermal control subsystem for the 7 kWe design was assumed to be a scaled down version of the Space Station design, retaining the feature of one redundant loop.

The resulting 7 kWe design configuration is not necessarily consistent from a reliability standpoint, particularly for an unserviceable spacecraft. Future spacecraft needs will have an impact on power output conditions, redundancy, and weight and packaging limitations, which would tend to alter the design configuration from that assumed for the 7 kWe power system.

## 9.3 SILICON FLAT PLATE (PLANAR) SPACE ARRAYS

As the name suggests, a flat plate or planar array is one which consists of one or more flat panels, which when deployed from a spacecraft in orbit and pointed at the sun will directly convert incident solar energy into direct current power. The power level is directly proportional to the sun's intensity, the area of solar cells, and their solar electric conversion efficiency.

For example, the solar intensity as it reaches the proximity of earth has an annual average of 1371 $W/m^2$ (137.1 $mW/cm^2$). If a flat plate silicon solar panel contains one square meter of solar cell area whose efficiency is 14.5%, the panel would produce the following power:

$$1371\ W/m^2 * 1\ m^2 * 0.145 = 199\ W$$

Clearly, the above is oversimplified as a number of factors must be considered when estimating the power of a particular photovoltaic system, such as:

1. Cell packing factor (fraction of panel area covered by the solar cells)
2. Added area due to mast, containment box, dummy panels, tensioning system, etc.
3. Cell performance degradation in efficiency due to operation at a temperature greater than the standard rating temperature, and damage due to prolonged exposure of the cell and cover to radiation and ultraviolet (UV), plasma, and micrometeoroids
4. Electrical losses due to connecting the cells in a string, wiring harness, protective diode, and sequential shunt unit

Flat plate or planar arrays which have been powering spacecraft since the early 1960s have been of two generic types: rigid and flexible, the latter only reaching technical maturity and flight qualification status on a limited scale in 1971 on the Air Force FRUSA program, and on the current Space Station and Space Telescope programs. Rigid planar arrays are fabricated on sandwich panels constructed by bonding two thin facesheets to either side of a honeycomb core material while flexible blanket arrays use coated Kapton as the substrate material.

Most systems launched between the mid-1960s to the mid-1970s were designed around aluminum substrate technology wherein thin sheets, or facesheets, were thermovacuum bonded to either side of an aluminum hexagonal core similar to the sandwich construction used in aircraft and spacecraft structures. In this approach, film or pre-preg sheet adhesive is placed between the facesheets and the hex core, after which the thermosetting adhesive is reflowed either in a large oven using a vacuum bag or in an autoclave.

More recently, however, in response to the needs of space systems with higher power requirements and automatic deployment, array designers have gone to flexible blanket technology as a means of stowing large array wings accordian fashion. In both the cases of rigid and flexible arrays, the solar cell side of the array stacks are identical, the major differences being the stack of material representing the substrate or support structure.

As can be seen in figure 9.3-1, a protective cover glass filter is bonded to the face of each solar cell for the purpose of attenuating electrons and protons trapped in the earth's magnetosphere. During solar storms, and in periods of high solar activities, high energy protons invade the magnetosphere resulting in damage to the solar cells' P-N junction and a general diminishing of current generation. Currently, virtually every solar cell which is flying in space has the cover glass affixed to its surface using Dow Corning DC93-500 clear silicone, which is a two-part adhesive that meets spacecraft standards for vacuum weight stability and total volatile content. It also has a refractive index close to that of fused silica and is relatively stable under prolonged UV exposure.

The silicon solar cells used as models for this study are the same configuration as Space Station, having both the positive (P) and negative (N) electrical contacts on the bottom of the cells as a means of simplifying the attachment of the electrical interconnects. The interconnects can then be laminated directly into the Kapton blanket substrates in the form of thin flexible copper ribbons and soldered to the solar cell contacts on the bottom side only after which the cell circuits are bonded to the Kapton substrates.

At the array wing level, a wing is subdivided into two blankets each consisting of a number of "panels" which are attached together at hingelines to form a common blanket assembly capable of being folded and unfolded like an accordian. During launch, the blankets are folded into stowage containers, and when they reach the proper deployment mode the container is unlatched and the blankets are extended to their full length by an Astromast, which is described in figure 9.3-2.

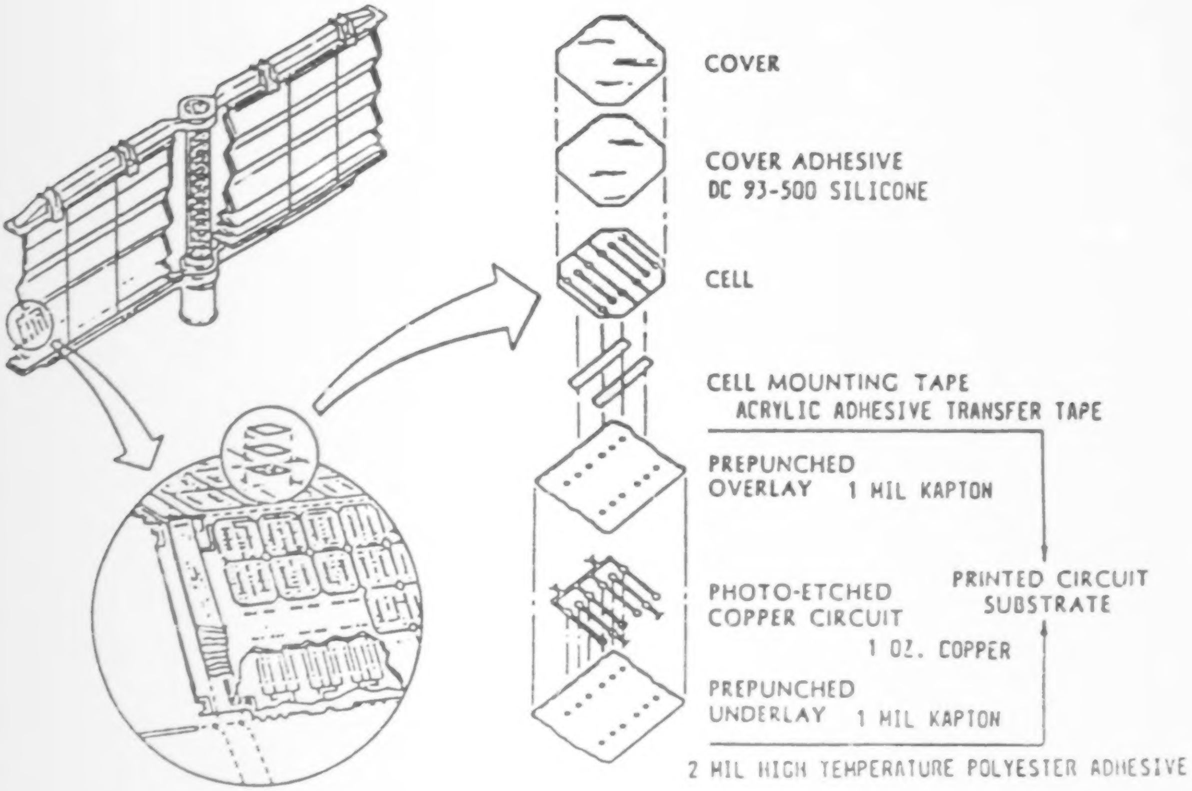
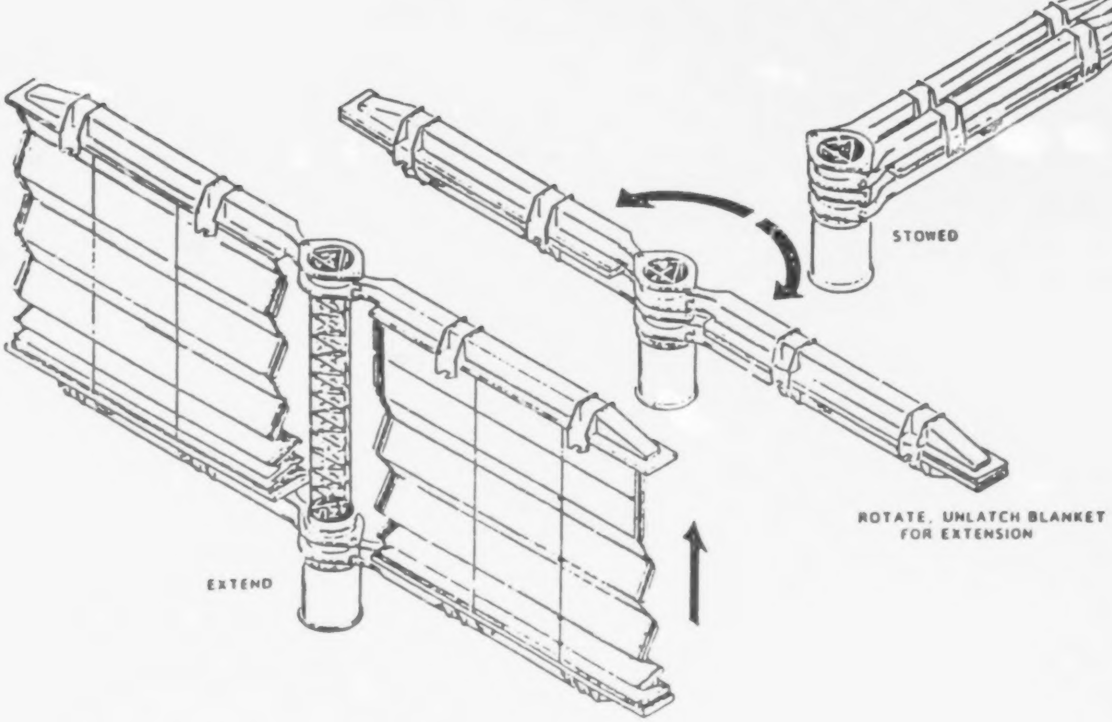
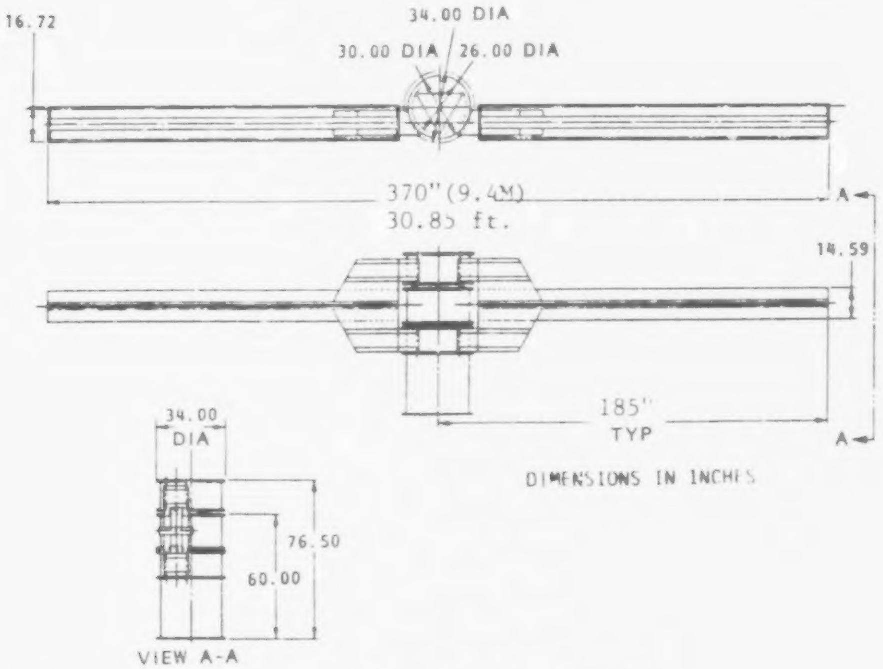

COVER
COVER ADHESIVE
DC 93-500 SILICONE
CELL
CELL MOUNTING TAPE
ACRYLIC ADHESIVE TRANSFER TAPE
PREPUNCHED
OVERLAY 1 MIL KAPTON
PHOTO-ETCHED
COPPER CIRCUIT
PRINTED CIRCUIT
SUBSTRATE
1 OZ. COPPER
PREPUNCHED
UNDERLAY 1 MIL KAPTON
2 MIL HIGH TEMPERATURE POLYESTER ADHESIVE

Figure 9.3-1. Exploded View of Photovoltaic Array

During eclipses the spacecraft solar power system undergoes a variety of rather extreme temperature excursions. These large-area photovoltaic wings act as their own radiators and do not remain as flat surfaces, instead curving and deflecting to one degree or another depending on the properties of the materials from which they are constructed and the thermal gradients through their cross sections. Because of this, the polymers used to bond the solar cell circuits to the substrates are not only adhesives but stress attenuators, and must exhibit low elastic moduli throughout a wide temperature range, as well as being applied and cured with a thicker cross section than normal adhesive bondlines. Additionally, like all adhesives used on spacecraft, these bonding polymers must be "clean" in vacuum, free of volatiles, and volumetrically stable over a long mission duration.


STOWED
ROTATE, UNLATCH BLANKET FOR EXTENSION
EXTEND

STORED CONFIGURATION


16.72
34.00 DIA
30.00 DIA
26.00 DIA
370" (9.4M)
30.85 ft.
A
14.59
185"
TYP
A
34.00
DIA
76.50
60.00
VIEW A-A

DIMENSIONS IN INCHES

Figure 9.3-2. Solar Array Wing

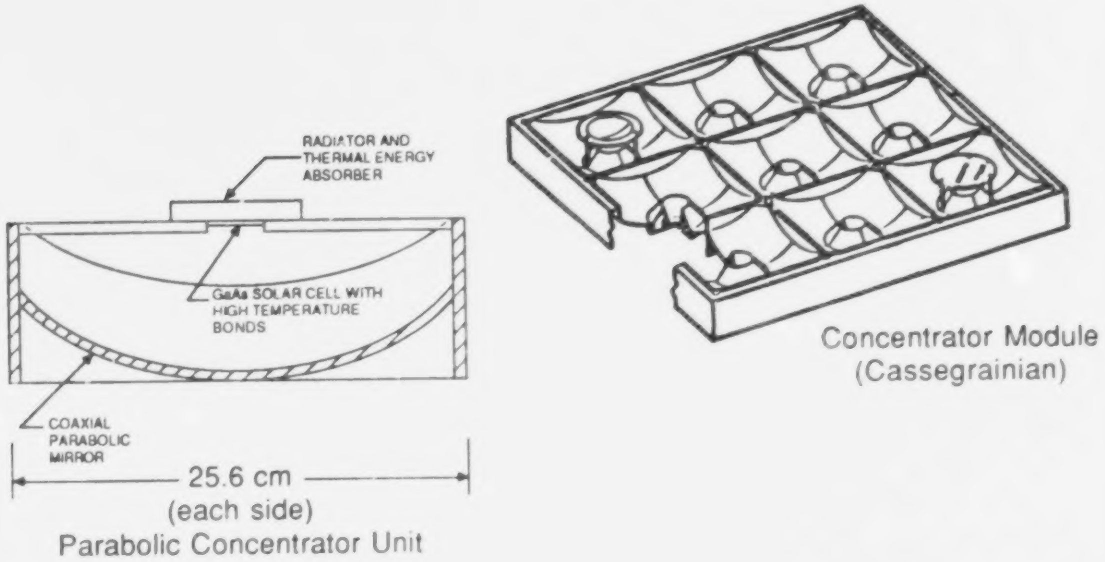
Each wing assembly must look at the sun continuously in order to maximize the power generation; this is accomplished by mounting the deployment container on a beta joint which is often referred to as a pointing gimbal, and is a motor-driven rotary coupling which receives signals from a sun-pointing control system and positions itself along the solar vector. This gimbal also contains slip or roll rings with which to transfer the solar electric dc power across the rotary joint and on into the spacecraft power conditioning and storage system.

## 9.4 GALLIUM ARSENIDE CONCENTRATOR SPACE ARRAYS

The gallium arsenide concentrator power system differs from a planar array in that the intercepted solar energy received by the opening (aperture) of the concentrator unit is concentrated optically to a higher flux density before it impinges on the GaAs solar cell and is converted to electrical power. The optical concentration is accomplished by a highly reflective parabolic mirror which not only changes the direction of the solar rays, but focuses them down to a much smaller target size and proportionately higher flux density.

For the Rocketdyne design concept considered herein (fig. 9.4-1), the ratio of input power to concentrated power is approximately 100, meaning that the input solar photon flux of 1371 $W/m^2$ is concentrated down to 137,100 $W/m^2$ as it strikes the GaAs solar cell, thus increasing the output of the individual solar cell approximately 100 times.

The advantages of this type of system are twofold; first, fewer solar cells are required to satisfy a given electrical power requirement, the advantage being that expensive solar cells are replaced with less costly concentrating hardware. Secondly, and more important, is the inherent hardenability of a concentrator due to the positioning of the solar cell which takes advantage of the radiation shielding afforded by the concentrating optics and reduces the Van Allen radiation degradation significantly. In the case of a fairly radiation-benign orbit, the shielding attenuates about 90% of the radiation degradation, but in a severe Van Allen environment the attenuation is only about 75% of that experienced by a comparable planar array.


RADIATOR AND THERMAL ENERGY ABSORBER
GaAs SOLAR CELL WITH HIGH TEMPERATURE BONDS
COAXIAL PARABOLIC MIRROR
25.6 cm
(each side)

Parabolic Concentrator Unit

Concentrator Module (Cassegrainian)

Figure 9.4-1. Gallium Arsenide Concentrator

It should be noted that a concentrator system operates at a higher temperature than a planar array and that a Cassegrainian concentrator unit (with secondary reflecting mirror) will operate at a lower temperature than a single-element parabolic concentrator as the cell radiator is located on the backside of the primary concentrator. This is important because solar cells gain efficiency at lower temperatures, but undergo less mission life radiation damage if they operate at higher temperatures, clearly indicating that the operating temperature design point must be a compromise between system efficiency and retained end-of-life (EOL) mission power.

## 9.5 PHOTOVOLTAIC ARRAY SIZING

The photovoltaic array power output must be greater than the nominal power system rating as a result of a portion of the orbit path being in eclipse. Additional energy absorbed during the sunlit period is stored electrically (batteries or regenerative fuel cells) so as to provide continuous power availability throughout the complete orbit. The array must be sized for the case of minimum solar input: the lowest seasonal solar intensity and the shortest sunlit orbit period. Losses for storage of electrical energy must also be included: charge, discharge, and electrical equipment for control of the charge/discharge. Various additional electrical losses related to the array sizing include sequential shunt unit (SSU), the main inverter unit (MIU), and various other electrical items in the circuitry. Power distribution and losses diagrams for 35 kWe and 7 kWe power systems are shown in figures 9.5-1 and 9.5-2.

The gross power requirement for the photovoltaic array can be calculated as follows:

$$PV = \frac{P}{E_{MIU} * E_{SSU}} \left[ \frac{1}{E_{PM-S}} + \frac{1}{E_{PM-E} * E_{ES}} * \frac{\text{maximum eclipse period}}{\text{minimum sun period}} \right]$$

Where:

| | |
|---|---|
| PV | array gross power |
| P | MIU power output |
| $E_{MIU}$ | MIU efficiency |
| $E_{SSU}$ | SSU efficiency |
| $E_{PM-S}$ | PMAD efficiency of direct use branch of circuit (active during sunlit portion of orbit) |
| $E_{PM-E}$ | PMAD efficiency of energy storage branch of circuit (active during sunlit portion of orbit) |
| $E_{ES}$ | BCDU and battery combined efficiency, defined as kWhr output ÷ kWhr input |

Array sizing results for the 35 kWe and 7 kWe power systems are as follows:

| | 35kWe | 7 kWe |
|---|---|---|
| P, kWe | 35.0 | 7.0 |
| $E_{MIU}$ | 0.910 | 0.910 |
| $E_{SSU}$ | 0.994 | 0.993 |
| $E_{PM-S}$ | 0.975 | 0.962 |
| $E_{PM-E}$ | 0.975 | 0.962 |
| $E_{ES}$* | 0.694 | 0.634 |
| Maximum eclipse period, min. | 35.68 | 34.72 |
| Minimum sun period, min. | 58.75 | 74.38 |
| PV, kWe | 74.4 | 13.98 |

*Note: Includes battery roundtrip efficiency of 0.780, and efficiencies of both the BCDU and the fault isolators.

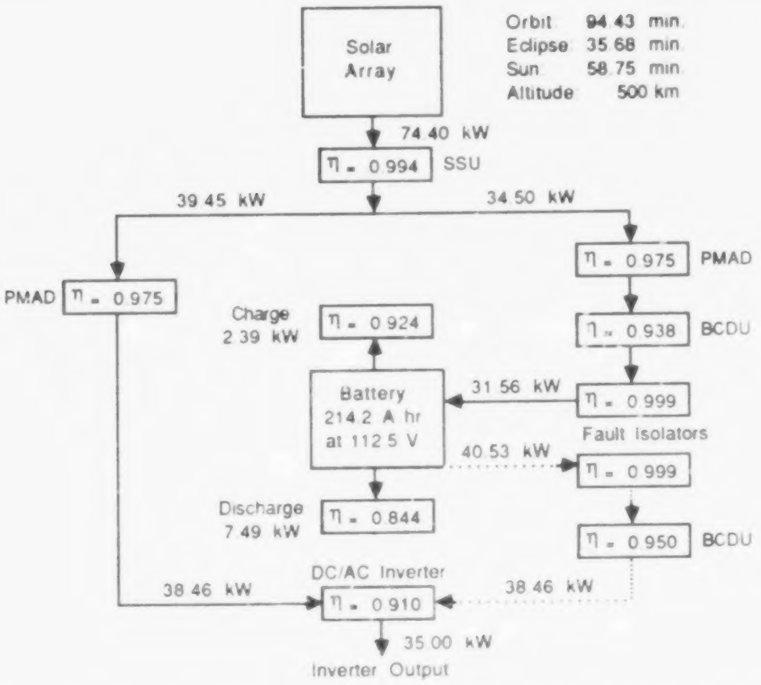

Solar Array
Orbit 94.43 min
Eclipse 35.68 min
Sun 58.75 min
Altitude 500 km
74.40 kW
η = 0.994
SSU
39.45 kW
34.50 kW
η = 0.975
PMAD
PMAD
η = 0.975
Charge 2.39 kW
η = 0.924
η = 0.938
BCDU
Battery 214.2 A hr at 112.5 V
31.56 kW
η = 0.999
Fault isolators
40.53 kW
η = 0.999
Discharge 7.49 kW
η = 0.844
η = 0.950
BCDU
DC/AC Inverter
38.46 kW
η = 0.910
38.46 kW
35.00 kW
Inverter Output

Figure 9.5-1. 35 kWe PV System Power Distribution and Losses - 500 km, 28.5° Orbit

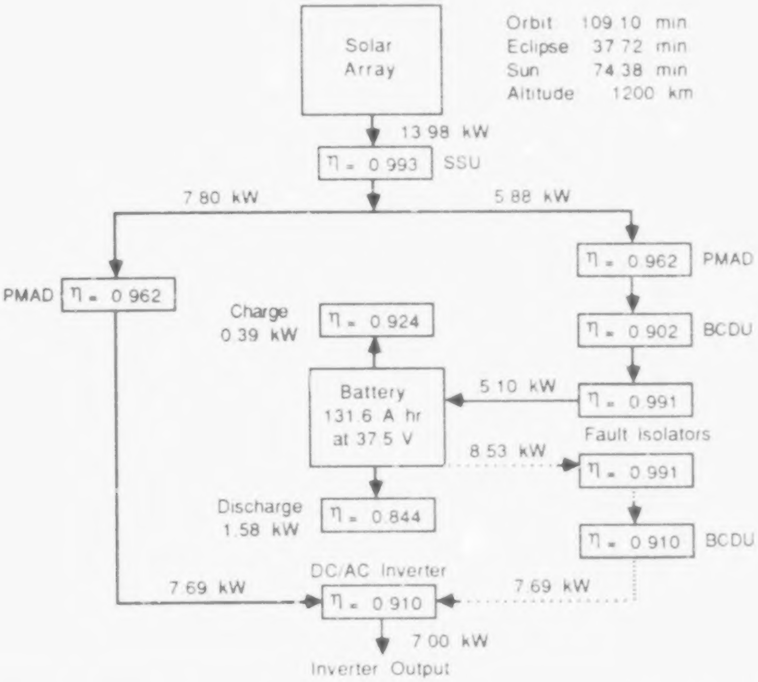

Solar Array
Orbit 109.10 min
Eclipse 37.72 min
Sun 74.38 min
Altitude 1200 km
13.98 kW
η = 0.993
SSU
7.80 kW
5.88 kW
η = 0.962
PMAD
PMAD
η = 0.962
Charge 0.39 kW
η = 0.924
η = 0.902
BCDU
Battery 131.6 A hr at 37.5 V
5.10 kW
η = 0.991
Fault isolators
8.53 kW
η = 0.991
Discharge 1.58 kW
η = 0.844
η = 0.910
BCDU
DC/AC Inverter
7.69 kW
η = 0.910
7.69 kW
7.00 kW
Inverter Output

Figure 9.5-2. 7 kWe PV System Power Distribution and Losses - 1200 km, 60° Orbit

### 9.5.1 Silicon Solar Array Sizing

Silicon solar cell performance for 7-year life for the orbit conditions of each of the design power conditions is presented in tables 9.5.1-1 and 9.5.1-2. The specified silicon solar cell efficiency of 14.5 was assumed to be at the conditions of 28C, AM0, 1-sun; cell operating temperature was assumed as 50C (323K) for both orbits. The natural Van Allen radiation degradation for each of the orbits was determined for the combined effects of electron and proton (dominant) fluence from reference 7. Cover glass thickness was chosen as 6 mil (the same as for Space Station) for both orbits. A 12 mil cover glass for the 1200 km, 60° inclination orbit would reduce the fluence by half; however, the additional weight did not warrant use of the thicker cover glass.

In order to design the array to operate at $\geq$160 V dc EOL voltage, the array blanket incorporates two panels connected in series for each 160 V dc power block. A number of power blocks are connected in parallel to produce the needed power. The total number of active panels in one blanket must therefore be multiple of two. The panels are patterned very closely after the Space Station panel design, which is 4.33 m by 0.389 m, each panel having four rows of 50 cells, and which includes a 0.1 m wide region on each end of the panel for the wiring harness. The solar cells are 8x8 cm square with corners missing so as to result in 60.14 $cm^2$ area. The panel and blanket sizing is an iterative procedure to select the number of cells per panel in multiples of four, and number of panels per blanket in multiples of two so as to result in an EOL voltage between 160-170 V dc and to choose that combination with the smallest excess amount of area.

For Space Station and for this study, each wing is composed of two blankets and a central extension mast. The blanket is made up of the chosen even number of panels plus two dummy panels, one at the bottom and one at the top of each blanket to allow for container shadowing and occlusion. Wing length is slightly longer than the blanket length to accommodate the halves of the containment box and a space for the tensioning and guidewire assemblies (~0.7 m). The extension masts designed for Space Station include a 0.71 m (28 in.) diameter mast for the station wings and a 0.48 m (19 in.) diameter

Table 9.5.1-1. 35 kWe, 500 km, 28.5° Orbit Silicon Cell EOL Power Analysis

| PARAMETER | | CORRECTION FACTOR | SHORT CIRCUIT CURRENT Isc, A | OPEN CIRCUIT VOLTAGE Voc, V | CURRENT AT MAXIMUM POWER Imp, A | VOLTAGE AT MAXIMUM POWER Vmp, V |
|---|---|---|---|---|---|---|
| BARE CELL* @ 28C | | X | 2.640 | 0.596 | 2.415 | 0.495 |
| COVERSLIDE | | x0.990 | 2.614 | X | 2.391 | X |
| I CIRCUIT LOSS | | x0.980 | 2.561 | X | 2.343 | X |
| V CIRCUIT LOSS | | -0.010 | X | 0.586 | X | 0.485 |
| RADIATION DEGRADATION | | | | | | |
| R (Imp) | | x0.973 | X | X | 2.280 | X |
| R (Isc) | FLUENCE | x0.973 | 2.492 | X | X | X |
| R (Vmp) | $1.18 \times 10^{13}$ | x0.988 | X | X | X | 0.479 |
| R (Voc) | | x0.937 | X | 0.578 | X | X |
| ENVIRONMENTAL | | | | | | |
| MICROMETEOROIDS | | x0.990 | 2.467 | X | 2.257 | X |
| PLASMA | | x0.990 | 2.443 | X | 2.235 | X |
| UV RADIATION | | x0.980 | 2.394 | X | 2.190 | X |
| THERMAL CYCLING | | x0.980 | 2.346 | X | 2.146 | X |
| HARNESS IR LOSS | | x0.970 | 2.275 | X | 2.082 | X |
| CONTAMINATION | | x0.980 | 2.230 | X | 2.040 | X |
| INTENSITY FACTOR | | x0.965 | 2.152 | X | 1.969 | X |
| OFF-POINTING | | x0.997 | 2.145 | X | 1.963 | X |
| TEMPERATURE @ 50C | | | | | | |
| I TEMP CORRECT (+0.101 $W/m^2C$) | | +0.013 | 2.158 | X | 1.976 | X |
| V TEMP CORRECT (-0.00205 V/C) | | -0.045 | X | 0.533 | X | 0.434 |
| 7 YEAR DESIGN-POINT TOTALS | | | 2.158 | 0.533 | 1.976 | 0.434 |

*NOTE: 8x8 cm CELL (60.14 $cm^2$), 14.5% BOL EFFICIENCY, SOLAR INTENSITY 1371 $W/m^2$

Table 9.5.1-2. 7 kWe, 1200 km, 60° Orbit Silicon Cell EOL Power Analysis

| PARAMETER | | CORRECTION FACTOR | SHORT CIRCUIT CURRENT Isc, A | OPEN CIRCUIT VOLTAGE Voc, V | CURRENT AT MAXIMUM POWER Imp, A | VOLTAGE AT MAXIMUM POWER Vmp, V |
|---|---|---|---|---|---|---|
| BARE CELL* @ 28C | | X | 2.640 | 0.596 | 2.415 | 0.495 |
| COVERSLIDE | | x0.990 | 2.614 | X | 2.391 | X |
| I CIRCUIT LOSS | | x0.980 | 2.561 | X | 2.343 | X |
| V CIRCUIT LOSS | | -0.010 | X | 0.586 | X | 0.485 |
| RADIATION DEGRADATION | | | | | | |
| R (Imp) | | x0.98 | X | X | 2.296 | X |
| R (Isc) | FLUENCE | x0.98 | 2.510 | X | X | X |
| R (Vmp) | $5.64 \times 10^{14}$ | x0.97 | X | X | X | 0.470 |
| R (Voc) | | x0.965 | X | 0.565 | X | X |
| ENVIRONMENTAL | | | | | | |
| MICROMETEOROIDS | | x0.990 | 2.485 | X | 2.273 | X |
| PLASMA | | x0.990 | 2.460 | X | 2.251 | X |
| UV RADIATION | | x0.980 | 2.411 | X | 2.206 | X |
| THERMAL CYCLING | | x0.980 | 2.363 | X | 2.162 | X |
| HARNESS IR LOSS | | x0.970 | 2.292 | X | 2.097 | X |
| CONTAMINATION | | x0.980 | 2.246 | X | 2.055 | X |
| INTENSITY FACTOR | | x0.965 | 2.167 | X | 1.983 | X |
| OFF-POINTING | | x0.997 | 2.161 | X | 1.977 | X |
| TEMPERATURE @ 50C | | | | | | |
| I TEMP CORRECT (+0.101 $W/m^2C$) | | +0.013 | 2.174 | X | 1.990 | X |
| V TEMP CORRECT (-0.00205 V/C) | | -0.045 | X | 0.520 | X | 0.425 |
| 7 YEAR DESIGN-POINT TOTALS | | | 2.174 | 0.520 | 1.990 | 0.425 |

*NOTE: 8x8 cm CELL (60.14 $cm^2$), 14.5% BOL EFFICIENCY, SOLAR INTENSITY 1371 $W/m^2$

mast for the platforms. Both the 35 kWe and 7 kWe blanket designs result in fewer panels than for the POP (with 58 panels total, see figure 9.2-2); therefore, the smaller mast diameter was used for wing sizing and weight for this study. The results of the silicon solar array sizing are shown in table 9.5.1-3.

Table 9.5.1-3. Silicon Solar Array Sizing Summary

| | 35 kWe | 7 kWe |
|---|---|---|
| Array EOL power, kWe | 74.40 | 13.98 |
| Number of wings per array | 4 | 1 |
| Number of blankets per wing | 2 | 2 |
| Blanket EOL power, kWe | 9.30 | 6.99 |
| Cell operating temperature, C (K) | 45 (318) | 45 (318) |
| Cell current at maximum power, A | 1.975 | 1.989 |
| Cell voltage at maximum power, V | 0.435 | 0.426 |
| Number of cells per panel | 196 | 200 |
| Panel packing factor(1) | 0.71 | 0.713 |
| EOL voltage, two panels, V(2) | 168.5 | 168.4 |
| Power per panel, W | 166.4 | 167.5 |
| Required panels per blanket | 55.9 | 41.7 |
| Active panels per blanket (even number) | 56 | 42 |
| Total panels per blanket(3) | 58 | 44 |
| Margin of power at EOL, % | 0.2 | 0.6 |
| Blanket width, m | 4.25 | 4.33 |
| Blanket length, m | 22.6 | 17.1 |
| Blanket active area, $m^2$ | 96.0 | 74.2 |
| Wing width (with mast), m | 10.0 | 10.2 |
| Wing length, m | 23.3 | 17.8 |
| Wing sail area, $m^2$ | 233 | 182 |
| Array sail area, $m^2$ | 933 | 182 |
| Active panel weight (each), kg | 1.758 | 1.794 |
| Blanket panel weight, kg | 98.5 | 75.3 |
| Electrical harness weight, kg | 13.4 | 10.8 |
| Blanket box, tensioning, and latch assembly weight, kg | 41.0 | 37.6 |
| Total wing weight, kg | 380 | 303 |
| Two blankets, kg | 306 | 247 |
| Atomic oxygen protection, kg | 12 | 0 |
| Mast, cannister, and drive assembly, kg (4) | 62 | 56 |
| Total solar array weight, kg | 1522 | 303 |
| Sequential shunt unit weight, kg | 56 | 14 |

Notes:

1. Cell area ÷ panel area.
2. Diode loss per panel of 1 V dc.
3. Includes two dummy panels.
4. 0.48 m (19 in.) diameter mast.

## 9.5.2 Gallium Arsenide Concentrator Solar Array Sizing

Gallium arsenide solar cell performance for 7-year life for the orbit conditions of each design power level is as follows:

| | 35 kWe | 7 kWe |
|---|---|---|
| Orbit altitude, km | 500 | 1200 |
| Orbit inclination, deg | 28.5 | 60 |
| Combined radiation fluence | $1.18 \times 10^{13}$ | $5.64 \times 10^{14}$ |
| Fluence at 85% attenuation* | $1.77 \times 10^{12}$ | $8.46 \times 10^{13}$ |
| Cell average operating temperature, C (K) | 80 (353) | 80 (353) |
| Concentrator unit power, W | 13.31 | 12.65 |

*Note: Assumed attenuation due to geometric shielding by the concentrator hardware.

Array sizing was based on concentrator units of 25.6 cm by 25.6 cm square (see figure 9.4-1), with a packing factor of 0.9 assumed for the array. The incoming solar energy is concentrated 100 times by the coaxial parabolic mirror before impinging on the GaAs solar cell mounted at the focal plane of the concentrator. The cell is thermally coupled to a circular radiator designed to maintain the cell at an average temperature of 80C (353K) while illuminated. The concentrator array sizing results are presented in table 9.5.2-1.

Table 9.5.2-1. GaAs Concentrator Solar Array Sizing Summary

| | 35 kWe | 7 kWe |
|---|---|---|
| Array EOL power, kWe | 74.40 | 13.98 |
| Number of wings per array | 4 | 1 |
| Number of concentrator units | 5590 | 1105 |
| Concentrator unit area, $m^2$ | 0.0655 | 0.0655 |
| Total area of concentrator units, $m^2$ | 366 | 72.4 |
| Array area (at 0.9 packing factor), $m^2$ | 407 | 80.5 |
| Concentrator unit weight, kg | 0.196 | 0.196 |
| Total weight of concentrator units, kg | 1096 | 217 |
| Electric harness weight, kg | 54 | 11 |
| Atomic oxygen protection, kg | 48 | 0 |
| Structure and deployer weight, kg | 251 | 55 |
| Total solar array weight, kg | 1449 | 283 |
| Sequential shunt unit weight, kg | 56 | 14 |

The concentrator arrays include structure, wiring harness, and atomic oxygen protection (for the 500 km orbit). The array structure was selected as the Astro Extendible Support Structure (ESS) baseline design (ref. 8), adjusted for both array area and array area density (2.9 kg/m$^2$ for this study versus 5.7 kg/m$^2$ for reference 8). Deployer weight was 17 kg. An estimated 7 kg was included for each wing for structure between the ESS and the array gimbal joint. Structure and deployer weight for each wing was estimated as: 0.38 * array area + 24 kg. The concentrator array design as regards launch packaging, deployment, details of structure, actual packing factor versus estimated, etc., was not pursued to the same level of detail with which the silicon array was designed. Comparing the two types of array design, it can be seen that the GaAs concentrator arrays are about the same weight as the silicon arrays, but are considerably smaller.

## 9.6 PHOTOVOLTAIC POWER SYSTEM WEIGHT SUMMARY

A weight summary for the four photovoltaic systems described above is presented in table 9.6-1.

Table 9.6-1. Photovoltaic Power System Weight Summary

| | 35 kWe | 7 kWe |
|---|---|---|
| Power management and distribution*, kg | 1339 | 448 |
| Batteries, kg | 2373 | 571 |
| Thermal control, kg | 1633 | 464 |
| Subtotal | 5345 | 1483 |
| | | |
| Silicon planar array, kg | 1578 | 326 |
| Total, silicon array power system, kg | 6923 | 1800 |
| | | |
| GaAs concentrator array, kg | 1505 | 297 |
| Total, GaAs array power system, kg | 6850 | 1780 |

*Note: SSU weight included with array weight.

# 10.0 CRITICAL DEVELOPMENT AREAS FOR SOLAR DYNAMIC POWER SYSTEMS

The work reported in this section corresponds to Task IV from the SOW.

## 10.1 OBJECTIVE

Critical development areas shall be identified and recommendations made where there is potential for improvement for each power system (Brayton, Rankine, and Stirling). Advanced concepts may be considered. Performance improvement areas shall include, but not be limited to, the following:

1. Increased efficiency (system and components)
2. Reduced area (solar collector, radiator)
3. Increased reliability (system and components)
4. Reduced mass (system and components)
5. Increased life (resistance to degradation)
6. Reduced complexity
7. Technology readiness

## 10.2 IDENTIFICATION OF CRITICAL DEVELOPMENT AREAS

The three solar dynamic power system designs (CBC, heat pipe Stirling, and pumped loop Stirling) differ in PCU and receiver design but share common concentrator and radiator designs. The identification of critical development areas was divided into the following groupings:

1. Concentrator
2. Receiver/TES
3. PCU
4. Radiator
5. System

A listing of critical development areas within each grouping is presented in table 10.2-1, listed generally in the order of priority. The following sections expand upon these areas and identify where there is potential for improvement.

Table 10.2-1. Critical Development Areas

1.0 Concentrator
- 1.1 Configuration
- 1.2 Weight
- 1.3 Launch packaging
- 1.4 Deployment
- 1.5 Drag area
- 1.6 Losses
- 1.7 Degradation

2.0 Receiver
- 2.1 Primary and secondary heat pipes
- 2.2 TES integration - heat pipe receiver
- 2.3 Remote TES configuration (pumped loop)
- 2.4 TES material
- 2.5 TES containment
- 2.6 TES conductance enhancement
- 2.7 Aperture shield
- 2.8 Receiver shell
- 2.9 Losses
- 2.10 Weight

3.0 Power Conversion Unit
- 3.1 Performance
- 3.2 Stirling weight versus efficiency
- 3.3 Stirling sized for 35-40 kWe
- 3.4 Stirling heat pipe heater
- 3.5 Stirling temperature ratio to $\leq 3.0$
- 3.6 High temperature Stirling
- 3.7 High temperature Brayton
- 3.8 Stirling engine control

Table 10.2-1. Critical Development Areas (Concluded)

4.0 Radiator

- 4.1 Heat pipe selection (materials and working fluid)
- 4.2 Drag area

5.0 System

- 5.1 Excess heat rejection
- 5.2 Fabricability
- 5.3 Ground test
- 5.4 Reliability/fault tolerance
- 5.5 Computer software for design optimization

Each of the following sections presents a discussion of the developmental issues of major concern within the subject grouping (concentrator, etc.). Each section also includes a table with the complete listing of critical development issues as were listed in table 10.2-1. Each table is organized in a manner similar to a work breakdown structure by making brief notation of the many factors and concerns to be considered within the particular critical development area.

The critical development issues discussed in section 10 are further amplified in section 11 wherein detailed advanced technology tasks are outlined; the tasks presented in section 11 provide a means of addressing the critical technology issues.

## 10.3 CONCENTRATOR CRITICAL DEVELOPMENT AREAS

A listing of issues which must be considered in identification of the concentrator critical development areas is presented in table 10.3-1. The preferred design would be a concentrator which would be very light weight, would provide long term high efficiency (reflectivity, intercept factor due to surface accuracy and pointing, and reflective surface packing factor), would provide durability to atomic oxygen and micrometeoroids, and which could be easily deployed. Such a design would be universal to all missions. Present designs such as the truss hex, splined radial panel (SRP), and Fresnel lens each have features falling short of the universal concentrator such that concentrator design selection is presently quite mission dependent.

### 10.3.1 Mission Considerations

The truss hex concentrator chosen for Space Station is suitable for LEO manned or mantended operation where drag area, atomic oxygen tolerance, on-orbit replaceability, and technology risk were more important issues than minimum weight. Furthermore, Shuttle launches of the Space Station hardware were expected to be more volume limited than weight limited. Added weight on Space Station decreases the drag-to-weight ratio (ballistic coefficient), which has a positive effect upon time between altitude reboosts.

Table 10.3-1. Concentrator Critical Development Issues

1.0 Concentrator

- 1.1 Conceptual designs
  - 1.1.1 Different designs for different missions
    - Launch weight limited
    - Launch volume limited
    - Lifetime
    - Environment
    - Serviceable versus unserviceable
    - Configurations
      - Truss hex
      - Splined radial panel
      - Domed Fresnel
      - Other
    - No single design best for all missions
  - 1.1.2 Design tradeoffs and analysis
    - Support structure
    - Reflective surface design
    - Insulation
      - Minimize distortion from thermal cycling
    - Deployment and restow
    - Scalability
    - Producibility
    - Receiver compatibility
    - Requirements imposed on system
      - Pointing accuracy
    - Development risk
  - 1.1.3 Rank conceptual designs
    - Optimization mission dependent
    - Assess penalty for alternate concentrator designs
    - Include input from all concentrator (1.0) subtasks
      - Weight, packaging, deployment, etc.

Table 10.3-1. Concentrator Critical Development Issues (Continued)

- 1.1.4 Mechanical design
  - One or more design configurations
  - Full-scale flight weight design (35-40 kWe size)
  - Ground calibration method and tooling design
- 1.1.5 Fabrication and test
  - Establish need and objectives
  - Fabricate one or more design configurations
  - Ground calibration
  - Ground test
    - Deployment and restow
    - Performance
  - Establish need for flight test
    - Flight test program

1.2 Weight

- 1.2.1 Survey typical missions through ~2010
  - Weight limited missions
  - Benefits of weight reduction: cost, etc.
- 1.2.2 Weight reduction design study
  - Conceptual design configurations
  - Impact on performance, cost, and launch volume
- 1.2.3 Rank conceptual designs
  - Weight limited missions

1.3 Launch Packaging

- 1.3.1 Survey typical missions through ~2010
  - Volume limited missions
  - Benefits of volume reduction: cost, etc.
- 1.3.2 Launch volume reduction design study
  - Conceptual design configurations
  - Impact on performance, cost, and weight
- 1.3.3 Rank conceptual designs
  - Volume limited missions

Table 10.3-1. Concentrator Critical Development Issues (Continued)

1.4 Deployment
  1.4.1 Survey missions for deployment requirements
    - Self-deployed
    - Robotic deployment/assembly
    - Man-assisted deployment/assembly
  1.4.2 Identify restow and disposal requirements
  1.4.3 Design study
    - Conceptual design configurations
      - Without restow
      - With restow
    - Impact on weight, launch volume, cost, and performance
1.5 Drag area
  1.5.1 Survey missions for low-earth orbits
  1.5.2 Apparent drag area
    - Compared to aperture area (area ratio)
    - For different concentrator configurations
    - Dependency on orbit
      - Orbit inclination
      - Altitude
1.6 Losses
  1.6.1 Receiver intercept factor
    - Surface slope error
    - Reflected/refracted beam non-specularity
    - Pointing error
    - Segmented surface approximation to a parabolic surface
  1.6.2 Trade concentrator and receiver losses
    - Receiver reradiation and reflection
    - Optimize weight of concentrator plus receiver
      - Weight variance with each error
    - Target values for errors
      - Current technology

Table 10.3-1. Concentrator Critical Development Issues (Concluded)

1.6.3 Error reduction trade
- Penalties for error reduction
  - Cost - development and production
  - Weight
- Target values for errors
  - For A/D concentrators

1.6.4 Recommended development program
- Establish improvements needed
  - For different A/D concentrator configurations
- Component and subassembly tests required

1.7 Degradation

1.7.1 Mission survey as to exposure to environment

1.7.2 Degradation mechanisms
- Atomic oxygen
- Ultraviolet and infrared
- Micrometeoroid and debris
- Contamination
- Thermal cycling

1.7.3 Evaluation of SOA material deficiencies
- Both surface reflectance and specular transmittance (Fresnel)
- Penalty imposed with SOA materials
- Cost of periodic upgrading (replacement)

1.7.4 Recommended development program
- Establish improvements needed
  - For different A/D concentrator configurations
  - Prioritize areas of improvement as to cost versus benefit
- Materials testing
  - Reflective/refractive materials
  - Surface coatings
  - Bonding materials
  - Probable contaminants

1.7.5 Suitability of concentrator types for different environments

Concentrator issues for higher altitude orbits are significantly different than for LEO, as the concerns of drag and atomic oxygen lose significance, and as the concerns of weight and high reliability (unserviceable spacecraft) come to the fore. At higher altitudes, area alone is not an issue, nor are reflectivity and surface accuracy individually important. For example, if a concentrator has 10% lower efficiency including intercept factor, but weighs 50% less than a competing design, then it is most probably much superior from a power system standpoint. Therefore, what design is judged best must be measured in terms of the mission and in terms of overall power system performance.

### 10.3 2 Comparison of Concentrator Concepts

Data from reference 3 for the truss hex, SRP, and Fresnel lens concepts was compared on the basis of energy absorbed by the solar receiver. The study included concentrator losses, intercept factor, and receiver reflective and reradiation losses. The results, expressed as weight and stowed volume per unit of energy absorbed (for a nominal energy level of 185±20 kWt were:

| | Specific Weight kg/kWt | Stowed Volume $m^3$/kWt |
|---|---|---|
| Truss hex | 4.1 | 0.146 |
| Splined radial panel (SRP) | 1.37 | 0.042 |
| Fresnel lens | 1.4-2.0 | 0.029 |

which included estimated strut weight but not volume. To each of these must be added controls and sensors of perhaps 30 kg.

The SRP and Fresnel designs indicated above were sized for Space Station organic Rankine application with much lower receiver temperatures. The designs would have to be reevaluated for higher temperature and higher concentration ratio application as it impacts both facet sizing and concentrator/receiver losses. The Fresnel lens concept appears to require a low concentration ratio and quite possibly alternate receiver design concepts due to the pattern of the concentrated energy rays. This is as a result of spreading of the concentrated rays due to refraction, which is most pronounced at the outer edge of the Fresnel lens.

The results of this study place the baseline truss hex concentrator weight at 14-19% of power system weight for the CBC and Stirling designs. There appears to be the potential for very substantial weight savings with other concepts; however, the concepts require further engineering development for high temperature application. Plans for investigation of other, even lighter weight concepts are included in the NASA ASD Program activities.

## 10.4 RECEIVER/TES CRITICAL DEVELOPMENT AREAS

A listing of issues which must be considered in identification of the receiver/TES critical development areas is presented in table 10.4-1. The function of the receiver is to absorb the concentrated solar energy, and that of the TES is to store adequate energy for continuous operation during the eclipse portion of an orbit. TES charging requires that the receiver and concentrator both be oversized so as to capture sufficient energy during the period of available sunlight to provide continuous power generation throughout the entire orbit.

The latent heat method of thermal energy storage was chosen for this study, in common with the approach taken for the proposed Space Station designs. The choice between latent heat versus sensible heat storage continues to be studied for solar dynamic space power generation, with significantly different design and operation issues associated with each. These issues are reflected in receiver/TES size and weight, variability of engine inlet temperature, and control of the power conversion unit. For the latent heat designs, the weight of the TES material amounts to 4-6% of power system weight (for LiF); however, it is the additional weight required for containment and for enhancing heat transfer within the TES that adds greatly to receiver/TES weight. The NASA ASD Program is addressing the issue of advanced receivers through contracts with Garrett AiResearch Division of Allied Signal and Sanders Associates.

### 10.4.1 Thermal Energy Storage

Results of this study indicate that LiF is a preferred material for TES, for the design concepts investigated. This is as a result of the greater

Table 10.4-1. Receiver/TES Critical Development Issues

2.0 Receiver/TES

2.1 Primary and secondary heat pipes

2.1.1 Preliminary design (heat pipe receiver)

- Heat pipe/TES configuration
- Wick configuration
- Design for 1-g and 0-g operation

2.1.2 Working fluid selection

- Analysis

2.1.3 Heat pipe material selection

- Analysis
- Laboratory tests (HP fluid compatibility)
  - Coupons
  - Stressed samples

2.1.4 Individual heat pipe performance testing

- Single or few-pipe test setup
  - Full scale heat pipes
- Test different inclination angles
  - Capability of 1-g operation
- Nominal heat transfer rates
- Nominal heat pipe operating temperatures
- Increased heat transfer rates
  - Rate 1.5-2.0 times nominal
  - Test primary heat pipe burnout limits

2.1.5 Test full-scale heat pipe receiver

- Compatibility with 1-g environment
  - Analytical verification before proceeding
  - Utilize single pipe test of inclination angles
- Construct complete heat transport assembly
  - Primary heat pipes
  - TES
  - Secondary heat pipes

Table 10.4-1. Receiver/TES Critical Development Issues (Continued)

- Test in vacuum
  - Infrared heating or other
  - Moveable heat lamp assembly
    - Simulate sunlight/eclipse intervals
  - Simulate continuous heat rejection
- Flight weight receiver shell
  - Desirable, not required

2.2 TES integration - heat pipe receiver

2.2.1 Containment material selection

- Analysis
- Laboratory tests
  - Coupons and stressed samples
  - Detailed in TES containment (2.5)

2.2.2 Conductance enhancement

- Analysis
- Laboratory tests
  - Performance
  - Material compatibility
  - Detailed in TES conductance enhancement (2.6)

2.3 Remote TES configuration (pumped loop receiver)

2.3.1 Containment material selection

- Analysis
  - TES cannisters
  - Containment vessel
- Laboratory tests
  - Coupons and stressed samples
  - Detailed in TES containment (2.5)

Table 10.4-1. Receiver/TES Critical Development Issues (Continued)

2.3.2 TES cannister freeze/thaw cycle

- Analysis of freeze/thaw cycle
- Laboratory testing of freeze/thaw cycle
  - Full-scale cannisters
  - TES material without additives
  - TES material with liquid metal additive
  - TES material with metallic foam or felts
  - Void location in frozen cannisters
  - Recommended configurations
- Cannister diameter
  - Analysis for diameters over 1 inch
  - Recommended diameter
- Long duration testing freeze/thaw cycle
  - Recommended configurations and diameter
    - Several cannisters each configuration
  - Long duration testing
    - Deterioration of TES, additive, or cannister
  - Thermal performance
    - Analytical prediction and correlation
    - Test results

2.3.3 Conductance enhancement

- Analysis
- Laboratory tests
  - Performance
  - Material compatibility
  - Detailed in TES conductance enhancement (2.6)

Table 10.4-1. Receiver/TES Critical Development Issues (Continued)

2.4 TES material

- 2.4.1 Material selection
    - Baseline - LiF salt
        - Superior heat of fusion
    - Alternate materials
        - Eutectic salt mixtures
        - Silicon/metal eutectic
    - Basic laboratory research
        - Forming the eutectic
        - Density, melting temperature, heat of fusion
            - Narrow melting temperature range
        - Thermal conductivity, expansion coefficient
        - Containment materials compatibility
- 2.4.2 Comparison of alternative TES materials
    - Analytical comparison
        - Basis of power system weight
    - Recommend TES alternates for further development

2.5 TES containment

- 2.5.1 Containment material compatibility
    - Design and analysis
        - TES containment configurations
    - Compatibility with TES materials
    - External exposure
        - Heat pipe fluid (liquid metal and vapor)
        - Heat transport fluid (liquid metal)
    - Laboratory tests
        - Coupons
        - Stressed samples
        - Exposure to TES material and liquid metal
        - Susceptibility to TES material impurities

C-3

Table 10.4-1. Receiver/TES Critical Development Issues (Continued)

- 2.5.2 TES material purity
    - Analysis
        - Available TES materials
        - Typical impurities found
        - Methods of refinement
        - Containment material tolerance to impurities
    - Laboratory tests
        - Methods of refinement
        - Containment material tolerance
- 2.5.3 Joining
    - Weld qualification procedures
    - Testing
- 2.5.4 TES containment filling
    - Filling procedures
    - Test verification

2.6 TES conductance enhancement

- 2.6.1 Analysis and correlation of test results
    - Enhanced heat flow through TES material
        - Finite element models
        - Mechanical conductance enhancement
            - Metallic fins or foam metal
            - Felts - metallic, nonmetallic, composites
        - TES additive for conductance enhancement
            - Alkali metal (Li with LiF, etc.)
    - Containment geometry
        - Diameter or section thickness
        - Segmented containment
    - Weight penalty
        - Fins, additive, and carnister increase
    - Weight reduction potential
        - Reduce TES margin
        - Larger TES cannister possible
        - Improve engine performance
    - Rank candidate enhanced configurations
        - Simulate effects upon power system

Table 10.4-1. Receiver/TES Critical Development Issues (Continued)

2.6.2 Test program
- Reference test with no enhancement
- Test candidate enhanced configuration
- Test TES configuration with liquid metal additive
- Correlation of test results with predictions

2.7 Aperture shield

2.7.1 Analysis and design
- Design requirement
  - Function of concentration ratio
  - Duration of exposure to beam
    - Rate of beam walk-off
  - Frequency of occurrence
  - Size of shield
    - Additional equipment to be shielded
  - Shield back-side heat transfer limitations
- Design approach
  - Heat management
    - Reradiation
    - Heat sink
    - Material wastage
  - Sensitivity to change in design requirements
  - Minimize weight

2.7.2 Material selection
- Potential for spacecraft contamination
- Development risk

2.7.3 Material testing
- Evaluate test environment required
  - Air versus vacuum testing
  - Concentration ratios up to 2000
  - Infrared source versus concentrated sunlight
- Choice of test facility
- Test program
  - Evaluate test results

Table 10.4-1. Receiver/TES Critical Development Issues (Concluded)

- 2.8 Receiver shell
    - 2.8.1 Analysis and design
        - Higher temperature operation
        - Materials selection
        - Conduction loss
        - Weight
    - 2.8.2 Material testing
        - Laboratory scale segment tests
            - Durability
            - Conduction loss
        - Vacuum test
            - High vacuum required
                - Foil surface emissivity
                - Interlayer gas conduction
- 2.9 Losses
    - 2.9.1 Analysis
        - Reradiation and reflection
            - Concentration ratio trade
        - Conduction
            - Receiver shell construction trade
        - Optimize power system weight
- 2.10 Weight
    - 2.10.1 Analysis
        - Primary optimization criterion
            - Perform at power system level
        - Major weight elements
            - TES material and containment
                - Conductance enhancement
            - Heat transport
                - Absorption and distribution
            - Shell and aperture shield
        - Trade study outputs to system level code
            - Interact with analysis and design tasks
        - Estimate cost payback for weight reduction
            - Indicates point of diminishing returns

benefits due to the high heat of fusion for LiF TES and lower reradiation loss associated with lower receiver temperature, as compared to the improvement in Carnot efficiency possible with higher temperature TES materials. With radical departure in receiver/TES design, it is possible that this conclusion could be reversed. Also, the search goes on for higher temperature TES media, such as silicon alloys.

The use of LiF or similar material for TES presents a number of problems. The volume changes by about 25% upon freezing, and the thermal conductivity of the material is quite low (with the liquid conductivity being approximately 30% that of the solid). These concerns result in reduced TES section thickness and encourage consideration of inclusions such as fins or felt metal within the TES material for conductance enhancement. The void creation upon freezing essentially requires that energy be withdrawn from a surface opposite the heated surface so that the void created by freezing during eclipse would be located adjacent to the surface to be heated during sunlight.

For the three receiver/TES conceptual designs developed for this study, the overhead weight associated with TES is as follows:

| 35 kWe | TES Material (LiF), kg | Containment* etc., kg | Total kg | Ratio: Total/TES |
|---|---|---|---|---|
| CBC | 266 | 625 | 891 | 3.35 |
| Heat pipe Stirling | 219 | 715 | 934 | 4.27 |
| Pumped loop Stirling | 221 | 885 | 1106 | 5.00 |

*Note: Includes felt metal for conductance enhancement: 20% dense for CBC and 16% dense for heat pipe Stirling.

The two Stirling receivers with external TES are much smaller than the CBC receiver with the TES located within the receiver shell. Comparing the total receiver weight, including TES, containment, heat absorbing tubes (for

Stirling), shell and aperture shield presents a more realistic comparison of receiver weights, as follows:

| 35 kWe | TES Material (LiF), kg | Total Receiver With TES, kg | Ratio: Total/TES |
|---|---|---|---|
| CBC | 266 | 1255 | 4.72 |
| Heat pipe Stirling | 219 | 1075 | 4.91 |
| Pumped loop Stirling | 221 | 1308 | 5.92 |

### 10.4.2 Conductance Enhancement

Two of the conceptual designs employ nickel felt metal within the TES volume which serves a two-fold purpose: thermal conductance enhancement, plus the felt metal acts as a wick to control location of the liquid and resultant void formed upon freezing. This technique has been shown to be quite effective (L.M. Sedgwick, Boeing, Contract NAS3-24669), but at the expense of considerable weight increase. For 16% dense nickel felt, the felt weighs 96% of the LiF weight assuming 3% void at the LiF melting point. Furthermore, containment volume must be increased by 19% to include the felt volume. Other felt materials should be examined, with possible candidates as follows:

| | Conductivity W/mK | Specific Gravity | Ratio |
|---|---|---|---|
| Nickel (baseline) | 74 | 8.88 | 8 |
| Nickel/copper, 40/60 by volume | 268 | 8.94 | 30 |
| Nickel/graphite, 40/60 by volume | 341 | 4.84 | 70 |
| Graphite | 520 | 2.15 | 242 |

Production techniques for very fine diameter nickel coated and copper coated fibers have been well developed in the last decade by American Cyanamid Company. The technique could possibly be applied to larger diameter fibers needed for metal felts. Looking at the tabulation above, essentially an order of magnitude improvement in the conductivity/density ratio would be possible with nickel/graphite fibers compared to nickel fibers, and even greater improvement with graphite fibers. NASA-LeRC is presently undertaking examination of the use of graphite fibers with LiF TES as part of their ASD Program.

Improvements in fiber conductivity and density pay off in reduced fiber weight, increased TES section thickness, and should substantially reduce containment weight due to reduced surface-to-volume ratio. Variation of PCU inlet temperature throughout the orbit can be reduced. Wicking concerns should be investigated further to establish what porosity would be acceptable for ground test (1-g) and whether that could be reduced for space hardware. The effect of surface tension and wetting of LiF in simple cannisters and cannisters with felt wicks is not well understood and should be analyzed and tested. Void formation under conditions of micro-gravity may well be controlled by the fluid properties and may not behave in the fashion presently anticipated.

### 10.4.3 Pumped Loop Stirling

For the pumped loop Stirling with remote TES, the preliminary design first prepared considered 1.22 m (48 in.) long, 2.54 cm (1 in.) diameter cannisters filled with LiF. At one point, addition of 10% by volume Li was considered with the anticipation that the LiF/Li might form a slush upon freezing. It was later found that the Li and LiF liquids were not miscible, and the solubility of Li in LiF liquid at the melting point was estimated as about 3%. It was not clear whether any Li would remain in solution with repeated freeze/thaw cycles. Incidental to this approach, it has been reported that the addition of a small amount of Li to LiF would probably be beneficial in removing impurities.

The concern with the pumped loop Stirling TES design is that the cannisters would freeze from one end to the other during eclipse, and then upon sunrise would being thawing from the first-frozen end (which is opposite the void location). This design approach worked well for the Rocketdyne IR&D test unit with LiOH cannisters, as the LiOH changes volume only slightly during freezing.

As a result of the perceived problem due to the LiF volume change, an alternative design approach would be required. One scheme would be to segment the long cannister into perhaps six individual cannisters and place baffles in the containment vessel such that the NaK would flow over the cannisters in a cross-flow pattern, causing freezing to occur in a radial rather than axial direction. The resultant void would be in the center of the cannister, and this would lead to stress loading of the cannister wall at the beginning of thaw due to the initial expansion of the melting TES material without communication with the central void volume. No firm resolution to the pumped loop Stirling TES design has emerged; the weights presented herein simply doubled the simple, single cannister weight to account for some eventual design resolution.

### 10.4.4 Receiver Shell

The receiver shell design was patterned after the Space Station CBC receiver shell, which was composed of a light weight formed insulation (ceramic fiber matrix) which forms the inner liner and serves as a mandrel for the multi-foil insulation (MFI), plus an aluminum outer shell. This same insulation composition was applied to the Stirling cycle TES (which is located external to the receiver), although it is probable that the TES insulation weight could be cut by perhaps 30% by reducing thickness of both the formed insulation and the aluminum shell. Regarding the receiver inner liner, the work on the A/D CBC receiver (L.M. Sedgwick, Boeing, Contract NAS3-24669) indicates that a metal liner is required to provide necessary reflection of energy for more uniform circumferential heating of the CBC TES tubes. Addition of a thin metal liner to the large size CBC receiver would increase receiver weight somewhat (about 30-40 kg). Neither of the Stirling receiver configurations would need a metal liner for energy reflection, as both the heat pipe and liquid metal pumped loop coil are very tolerant of heat flux maldistribution.

### 10.4.5 Aperture Shield

Another difference between reference 1 and the Boeing A/D CBC receiver is in the design of the receiver aperture shield, with the choice of graphite shield thickness being 1.27 cm (0.5 in.) and 5.08 cm (2.0 in), respectively. A compromise value of 2.54 cm (1.0 in.) was chosen for this study. As a result, the aperture shield plus a 0.127 cm (0.050 in.) aperture plate has a total weight which is equal to about half of the balance of the receiver shell weight for both the CBC and heat pipe Stirling designs. The pumped loop aperture shield is much larger and rectangular in shape so as to protect additional power system components along side of the receiver. More engineering work on the aperture shield could well adjust shield weight either up or down dependent on definition of the beam walk-off/walk-on requirements.

The graphite shield thickness is a function of both thermal properties and rate of material loss due to the concentrated beam passing across the aperture shield face (walk-off). In the event of loss of tracking control, the beam would walk off much more quickly in LEO orbit (due to the 94 minute orbit period) than for a terrestrial application such as the Boeing A/D ground-test receiver, or for a GEO application. An alternative design approach could be substitution of a very high temperature refractory metal arranged like shingles and built up in several thin layers ($\leq$0.025 cm) and perhaps backed with MFI insulation. For example, the density of tungsten is 9-10 times that of graphite; however, a total tungsten shield might weigh less than a graphite shield, and the tungsten shield would have indefinite life.

## 10.5 POWER CONVERSION UNIT CRITICAL DEVELOPMENT AREAS

A listing of issues which must be considered in identification of the power conversion unit (PCU) critical development areas is presented in table 10.5-1. The PCU includes thermodynamic conversion, electric power generation, and conversion of the electrical output to a common base of 20 kHz. The PCU also includes necessary controls including a parasitic load radiator (PLR) capable of dissipating the total power being produced.

### 10.5.1 Brayton Cycle

The CBC cycle proposed for this study is very similar to one of two concepts proposed for Space Station; however, operating at about 80K (145F) higher turbine inlet temperature. The critical issue for the CBC is operating experience at the higher temperature and realization of turbine and compressor performance and recuperator performance. All but the temperature issue will be dealt with under the Space Station program.

Table 10.5-1. Power Conversion Unit Critical Development Issues

3.0 Power Conversion Unit

- 3.1 Performance
  - 3.1.1 Electric power to user
    - Power conversion efficiency
      - User needs differ from alternator output
    - PCU efficiency
      - Alternator efficiency
      - Engine parasitic power
      - Engine conversion efficiency
  - 3.1.2 Effect upon power system
    - Size of major elements
      - Directly proportional to conversion efficiency
      - Concentrator, receiver/TES, radiator
    - Effects system weight and drag area
  - 3.1.3 State-of-the-art
    - Brayton cycle technology
      - Well developed
      - Improvement potential
        - Increased temperature ratio
        - Otherwise quite limited
    - Stirling cycle technology
      - Immature technology
      - Free piston Stirling engine at 35-40 kWe
        - Limited design and analysis
        - No test hardware or test data
      - Performance predictions uncertain
- 3.2 Stirling weight versus efficiency
  - 3.2.1 Engine trade studies
    - Efficiency gain diminishing with increase in engine weight
    - Design predictions 60-70% of Carnot efficiency
    - Engine with alternator generally 6-8 kg/kWe

Table 10.5-1. Power Conversion Unit Critical Development Issues (Continued)

3.2.2 Power system trade studies

- Engine efficiency effects size of major elements
  - Concentrator, receiver/TES, radiator
- Stirling power system predicted ~100 kg/kWe (gross)
  - Engine weight of low importance
  - Engine weight <10% of power system weight
- Design solar Stirling engine for high efficiency

3.3 Stirling sized for 35-40 kWe

3.3.1 State-of-the-art

- Detailed design maximum power
  - Sunpower 25 kWe Stirling Space Engine (SSE)
    - Single cylinder
    - Would produce ~35 kWe for solar conditions
- Maximum power tested
  - MTI 25 kWe Space Power Demonstrator Engine
    - Dual cylinder, 12.5 kWe each cylinder
    - Operation at design conditions not proven to date

3.3.2 Technology improvement

- Engine design and analysis
  - Single cylinder at 35-40 kWe
  - Design for solar power application
    - High efficiency ratio to Carnot
    - High temperature ratio
    - Similar thermal input as 25 kWe SSE
  - Heater configuration
    - Heat pipe heat transport
    - Pumped loop heat transport
  - Cooler configuration
    - Pumped liquid metal (NaK)
    - Pumped non-metallic fluid
    - Heat pipe feasibility

Table 10.5-1. Power Conversion Unit Critical Development Issues (Continued)

- Fabrication and test
  - Evaluate heat pipe heater versus pumped loop designs
    - Development risk
    - Impact on power system performance
      - Weight and area
  - Chose one or both for fabrication and test
  - Engine testing
- Correlation of Stirling codes
  - Compare test results with code predictions
  - Upgrade codes
  - Compare test results with design objectives
  - Utilize upgraded code to recommend design improvements
    - Design modification and test as appropriate

3.4 Stirling heat pipe heater

3.4.1 Conceptual designs of engine heater

- Power per heat pipe
- Coordinate with solar receiver/TES design
  - Equal number of heat pipes
- Integration with engine regenerator and cooler
- Fabricability

3.4.2 Analysis

- Heater heat transfer
- Engine performance versus pumped loop heater
  - Increased $T_H$ and $T_H/T_C$
    - Improvement limited by engine materials
  - Increased efficiency
  - Engine weight change
- Power system performance change
  - Size of major elements due to engine efficiency
  - Increased engine efficiency
    - Reduce size of major elements
  - Reduction in receiver/TES weight
    - TES reconfiguration with heat pipe receiver

Table 10.5-1. Power Conversion Unit Critical Development Issues (Continued)

- 3.4.3 Design recommendation
  - Design input into Stirling sizing (3.3)
- 3.5 Stirling temperature ratio to ~3.0
  - 3.5.1 Power system trade studies (Task II)
    - Minimum power system weight for $T_H/T_C$ near 2.6
    - Titanium/methanol radiator requires $T_H/T_C$ near 2.8 ± 0.1
    - Higher $T_H$ could result in $T_H/T_C$ ~3.0 for solar
      - Higher $T_H$ from heat pipe receiver/engine design
      - Higher $T_H$ choice due to higher temperature TES
  - 3.5.2 State-of-the-art
    - Engine analysis, design, and testing for nuclear application
      - $T_H/T_C = 2.0$
      - Unsuitable for solar application
  - 3.5.3 Technology improvement
    - Design and analysis at higher $T_H/T_C$
    - Design input into Stirling sizing (3.3)
- 3.6 High-temperature Stirling
  - 3.6.1 Power system trade studies (Task II)
    - Selected LiF as TES material for Stirling cycle
      - LiF melting temperature 1121K (1558F)
    - Higher temperature (ratio) improves Carnot efficiency
      - Continued research on higher temperature TES materials
  - 3.6.2 State-of-the-art
    - Stirling engines not tested at high temperature
      - Temperature corresponding to LiF TES

Table 10.5-1. Power Conversion Unit Critical Development Issues (Continued)

- 3.6.3 Technology improvement
  - Design and analysis for higher temperature ($T_H$)
    - Engine design
    - Materials selection
  - Design input into Stirling sizing (3.3)
  - Feasibility of operating existing engines at higher $T_H$
    - Recommend test program, if appropriate
- 3.7 High-temperature Brayton
  - 3.7.1 Power system trade studies (Task II)
    - Selected LiF as TES material for Brayton cycle
      - LiF melting temperature 1121K (1558F)
    - Higher temperature improves cycle efficiency
      - Continued research on higher temperature TES materials
      - Engine operation about 80K (145F) higher than proposed for Space Station CBC
  - 3.7.2 State-of-the-art
    - Numerous CBC components demonstrated
      - Near required operating temperatures
  - 3.7.3 Technology improvement
    - Design and analysis for higher temperature ($T_H$)
      - Up-rate from Space Station CBC design
      - Engine design
      - Materials
    - Fabrication and test
      - Demonstrate engine operation
        - Test bed operation
      - Performance
      - Endurance

Table 10.5-1. Power Conversion Unit Critical Development Issues (Continued)

3.8 Stirling engine control

3.8.1 Unsteady operating environment

- $T_H$ and $T_C$ vary about the orbit
  - May tend to be offsetting
- Design engine for constant power to user
  - Excess power may be wasted

3.8.2 Seasonal variations and orbital variations

- Design for nominal power for minimum solar input
- Excess energy input must be discarded
- Stirling power system design options
  - Dissipate excess heat from receiver
  - Bypass excess heat to engine radiator
  - Process excess heat through the engine
    - Increased power production
    - Reduced $T_H$, increased $T_C$
    - Reduced engine efficiency
    - Oversize alternator
    - Dissipate excess electrical energy

3.8.3 Design and analysis

- Conceptual designs for excess energy management
- Analysis
  - Transient response
- Concept tradeoffs
  - Weight
  - Reliability
  - Development risk

Table 10.5-1. Power Conversion Unit Critical Development Issues (Concluded)

3.8.4 Fabrication and test

- Depends on which is selected design
  - Dissipate excess heat from receiver
    - Breadboard test
    - Test with receiver
  - Bypass heat to engine radiator
    - Breadboard test
  - Process excess heat through engine
    - Breadboard test
    - Test with engine

### 10.5.2 Stirling Cycle

The results of this study indicate a potential for significant power system weight savings (18-22%) and similar area savings for the heat pipe Stirling versus the CBC power system. The technology maturity for the Stirling is behind that of CBC, with much more work needed on the Stirling to verify the performance, weight, and reliability of that engine. The NASA-LeRC sponsored 25 kWe SSE Program is designed to achieve this goal, although it is imperative that the differences between nuclear and solar Stirling application be included in that study.

The list of critical developments issues for Stirling is extensive, and will be dealt with in the aforementioned NASA SSE program. Chief among these will be: high temperature, performance of the heater/regenerator/cooler design, alternator performance, engine weight, hydrostatic bearing performance for the displacer and piston, engine control including start up/shut down and variable power conditions, and cycle analytical code development.

### 10.5.3 Excess Power Management

The solar system is normally designed for the condition of minimum solar input, resulting from longest eclipse and lowest intensity, such that the condition of maximum solar input results in excess energy which must be accommodated. The energy may be avoided or discarded before the engine, may be bypassed around the engine, or processed through the engine. For reasons of reliability, the latter option is usually preferred, which effects engine design and component sizing.

For the CBC, for example, bypassing some flow around the recuperator has the effect of quite directly reducing cycle efficiency resulting in increased power production and increased waste heat. For the Stirling engine, the approach would be to increase alternator applied voltage, which increases alternator power extraction. Increased power extraction from the TES can only occur by causing engine $T_H$ to decline (since the freezing/thawing of the TES approximates having a constant heat source temperature), such that more energy may be drawn from the receiver/TES. The higher energy through-put of the engine increases waste heat, increasing radiator temperature, which causes engine $T_C$ to rise. The reduced $T_H/T_C$ causes a reduction in Carnot efficiency; however, engine power output is higher.

Both the CBC PCU and Stirling PCU must be sized to accommodate minimum and maximum energy conditions. The hardware design and control methodology are being worked out for the Space Station CBC design, including the PLR needed to dissipate the excess electrical energy produced. The effect upon hardware and controls for the Stirling cycle must also be developed and tested as part of the NASA SSE program. Fortunately, degradation in engine operating efficiency is acceptable and desirable in the management of the excess energy for a solar power system; however, components such as the alternator must be appropriately oversized.

### 10.5.4 Electric Power Conversion

For purposes of this study, all PCU output, whether solar dynamic or photovoltaic, was converted to 20 kHz output so as to provide a common basis for comparison. In fact, many future spacecraft applications will require other power output conditions which may or may not necessitate conversion of the PCU electrical output. Space Station application will provide development of 20 kHz space-rated hardware. Should use of the military standard 400 Hz hardware be chosen for spacecraft application, then that equipment would have to be space-rated.

Cooling of the power conversion electronics should be examined for a minimum weight design. Optimization of the combined weight of the heat transport elements within the electronic converter (which carry the heat to the cold plate), and the weight of the cold plate and radiator, would probably results in an increase in cold plate temperature as compared to Space Station (designed for nominal electrical equipment baseplate temperature of 5±5C). Consideration could be given to combining this heat rejection with the engine waste heat radiator for CBC; however, a separate cooling circuit would be necessary for Stirling as the engine radiator loop would be much too hot. It may be desirable to use the same radiator panels for the separate cooling circuit as for the engine waste heat radiator.

## 10.6 WASTE HEAT RADIATOR CRITICAL DEVELOPMENT AREAS

A listing of issues which must be considered in identification of the waste heat radiator critical development areas is presented in table 10.6-1. Only the heat pipe type of radiator was considered for this study.

### 10.6.1 Dual-Slot Heat Pipe

The dual-slot radiator development by Grumman (ref. 2) has resulted in a weight reduction over the monogroove type of heat pipe. Furthermore, the cylindrical cross-section lends itself well to forming the heat pipes from tubing such as stainless steel, titanium, or other material which cannot be extruded, as is done for the aluminum monogroove configuration.

The radiator weight for the dual-slot design configuration, including panels for electronic components cooling, represents 25-30% of power system weight for the CBC and heat pipe Stirling power system designs developed for this study. The heat pipe panels comprise about 55% of the total radiator weight, the balance being the heat exchanger boom, panel attachment device, return line and structure.

For the heat pipe panels, most of the development and work has been done on the aluminum dual-slot. More work is required on fabrication of titanium tubes (forming capillary grooves and attachment of the baffle plate separating the liquid and vapor spaces), fabricating full length pipes, and the method of attachment of the aluminum fin to the titanium heat pipe. More development is required for the quick-disconnect concept, whereas the whiffletree clamping concept development is more mature. Both of these attachment techniques are for a serviceable application; further study of the folded heat exchanger boom concept could well show a weight advantage for an unserviceable application where panel replacement would not be a factor.

## Table 10.6-1. Radiator Critical Development Issues

4.0 Radiator

4.1 Heat pipe selection

4.1.1 Power system trade studies (Task II)

- Minimum power system weight conditions
  - Stirling engine temperature ratio $T_H/T_C = 2.6$
  - Radiator nominal inlet temperature 396K (254F)
  - Maximum inlet temperature about 40K (72F) higher
- Titanium/methanol heat pipes limited to about $\leq$260F
  - Nominal temperature ratio must be chosen >2.6
- 500 km, 28.5° inclination orbit
  - Stirling engine temperature ratio $T_H/T_C = 2.7$
  - Small increase in power system weight
  - 8-9% increase in radiator area
- 1200 km, 60° inclination orbit
  - Stirling engine temperature ratio $T_H/T_C = 2.9$
  - 2% increase in power system weight
  - 28% increase in radiator area

4.1.2 Higher temperature working fluid desirable

- Payoff for light weight concentrator
  - Reduced Stirling engine temperature ratio
- Potential for power system weight reduction
- Significant radiator area reduction
- Fluid selection
  - Water to about 475K (395F)
    - High vapor pressure and freeze point
    - Not compatible with stainless steel
  - Toluene 475K (395F) or higher
    - Low thermal performance
  - Other

Table 10.6-1. Radiator Critical Development Issues (Concluded)

- 4.1.3 Material selection
    - Metallic and non-metallic
    - Compatibility with working fluid
    - Impact properties
        - Susceptibility to damage from micrometeoroids and debris
    - Minimize specific weight (kg/kWt)
- 4.1.4 Fabrication and test
    - Selection of one or more higher temperature designs
        - Emphasis on minimum power system weight
    - Critical components design, fabrication, and test
        - Heat pipe evaporator heat exchanger
        - Performance testing
        - Long-term testing
    - Radiator design, fabrication, and test

4.2 Drag area

- 4.2.1 Sail area (planform)
    - Varies approximately as fourth-power of temperature
    - Predominantly heat pipe condenser area
    - Boom area
        - Structural member
        - Heat pipe evaporator heat exchanger
- 4.2.2 Radiator orientation
    - Tradeoffs
        - Edge-on to sun
        - Minimize drag integral
        - View factor of radiator and spacecraft components
            - High temperature radiator
        - Location relative to concentrator
            - Possible drag shielding
- 4.2.3 Ratio of drag area to sail area

### 10.6.2 Micrometeoroid Protection

Most of the heat pipe panel weight is in the pipes themselves, not the aluminum fin. Pipe wall thickness is almost entirely sized based on the micrometeoroid and debris hazard. That is why, for example, that the titanium/methanol wing panel weighs slightly less than the aluminum/ammonia monocoque panel (evaporator plus condenser weight divided by condenser area). The titanium wall thickness was 0.152 cm (0.060 in.) versus the aluminum wall thickness of 0.254 cm (0.100 in.) for the same survivability probability. (Note: The panel failure rates were compared on the basis of equal length, rather than the design lengths of 13.7 m (45 ft) for aluminum and 7.6 m (25 ft) for titanium.)

Alternative approaches to heat pipe micrometeoroid protection need to be examined; however, the rating still has to be kg/kWt for given conditions, so anything which degrades heat rejection must be properly accounted for. Examples of alternatives are:

1. No credit is taken on the wing configuration (see figure 6.2.1.6-1) for the presence of the 0.081 cm (0.032 in.) aluminum fin which covers a 180° arc of the pipe. This aluminum thickness is equivalent to about 0.05 cm of titanium, and the fact that they are bonded might change the impact mechanics. Since titanium and aluminum pipes appear to be about equally durable on a weight basis, then it would be advantageous to cover the other 180° arc of the heat pipe (in the wing configuration) with an additional 0.081 cm thick piece of aluminum. Then the titanium pipe wall thickness could be reduced by 0.05 cm to retain the same level of pipe durability. This change would reduce heat pipe panel (evaporator and condenser) by over 10%, reduce overall radiator weight by 6%, and power system weight by 1.2%.

2. The monocoque construction (see figure 6.2.1.6-1) likewise offers protection to the heat pipe in the form of the fin and the saddle, and a weight reduction for this design should also be possible.

3. Protect the heat pipes with material possessing higher impact protection, possibly a woven fiber material, either in direct contact with or in standoff configuration. The standoff offers better protection dynamics, but there is a penalty in a >15% loss of direct radiation area by shielding the pipe. The lower emissivity of a direct contact fiber protection material would also reduce performance somewhat.

4. Advanced composite non-metallic heat pipes are possible in the temperature ranges of the CBC and Stirling cycles. Such composites might prove to be both weight efficient and more durable to the micrometeoroid and debris hazards.

### 10.6.3 Heat Pipe Radiator Performance

For the power system designs developed and presented in this study, the radiator weight was found to contribute approximately 25-30% to the total power system weight. Grumman (ref. 2) developed several methods for attaching the panels to the heat exchanger boom, which are: the whiffletree clamp, a brazed attachment with quick disconnect, and the folded boom with brazed attachment (see figures 6.2.1.6-2 through 6.2.1.6-4). By using the whiffletree only for a double-sided boom configuration (where the whiffletree clamps two panels), this approach resulted in all three configurations weighting essentially the same. Comparing these with the Space Station pumped looped configuration (with 2 loops - one being redundant, and including a radiator automatic deployment mechanism) resulted in about a 16% weight reduction for the pumped loop configuration with the same thermal duty. The suggested change to reduce heat pipe wall thickness (section 10.6-2) could reduce heat pipe radiator weight by 6%. A comparison of radiators for 7 year life in an unserviceable situation requires 15% increase in area (and weight) for the heat pipe radiator (section 6.4.5), whereas designing a pumped loop radiator for 7 year life would result in a significantly larger weight increase.

Lacking any obvious means to reduce the weight of the radiator (which contributes 25% or more to power system weight), the only alternative appears to be reduction in radiator area. This was done throughout the trade studies, in the search for minimum system weight. Substitution of lighter weight concentrator and receiver/TES subsystems will cause the shift to smaller, hotter radiators for minimum system weight (section 6.4.4). However, the power system designs presented herein are presently constrained by the assumed maximum methanol operating temperature of $\leq$400K (260F).

### 10.6.4 Heat Pipe Working Fluids

The ammonia heat pipe operating temperature range of $\leq$350K (170F) is not suited for application to the CBC and Stirling power cycles. Methanol is generally satisfactory, with a low freeze temperature, and can be operated to about 400K (260F) before thermal decomposition becomes a concern. This temperature limits the power cycle operating temperature ratio. When light-weight concentrators and/or receiver/TES designs emerge, the methanol temperature limitation will prevent optimization at minimum power system weight as a result of increasing radiator temperature to achieve the minimum weight condition.

Two areas require further development: The upper temperature limit for methanol must be better defined; and the water heat pipe must be further pursued to establish what materials, in particular titanium, are compatible with water. Each of these requires long term testing to be conducted, which should be commenced promptly as such knowledge is already needed for the conduct of systems trades.

## 10.7 POWER SYSTEM CRITICAL DEVELOPMENT AREAS

A listing of issues which must be considered in identification of the power system critical development areas is presented in table 10.7-1. The system and component issues are in most cases inseparable insofar as achieving optimum designs. In turn, the power system is merely a subsystem of the host spacecraft. As technology changes, previous optimizations become obsolete and the process must be reiterated for the new technology. The systems approach must always be consulted in the quest for technology advancement of components and subsystems (as discussed in the previous segments of section 10), in order

Table 10.7-1. System Critical Development Issues

5.0 System

- 5.1 Excess energy management
  - 5.1.1 Quantity to be rejected
    - Excess heat available due to orbital variations
      - Seasonal
      - Orbit location
    - Assume fixed user power demand
      - Limits engine size
  - 5.1.2 Conceptual design and analysis
    - Avoid heat capture
      - Off-pointing
      - Concentrator aperture reduction
    - Rejection
      - From receiver/TES
      - Engine bypass to the engine radiator
      - Through the engine (reduced engine efficiency)
        - Oversize engine and radiator
        - Dissipate excess electric power
  - 5.1.3 Tradeoffs
    - System level impacts
      - Control strategy
      - Weight and area
      - Reliability
      - Development risk
    - Recommended design
  - 5.1.4 Fabrication and test
    - Breadboard test
      - Test program appropriate to receiver design
- 5.2 Fabricability
  - 5.2.1 Design phase requirement
    - Improve validity of trade studies
      - Feasibility of conceptual designs
      - Realistic weight and area values

Table 10.7-1. System Critical Development Issues (Continued)

5.2.2 Proof of production quality
- Features to be accommodated in design
  - Weld x-ray
  - TES fill
  - Heat pipe integrity
    - Structural
    - Performance
  - Clearances

5.2.3 Critical areas
- Stirling heat pipe receiver/TES/engine integration
- Stirling engine heaters and coolers

5.3 Ground test

5.3.1 Component, subsystem, system testing
- Testing an integral part of technology development
- Ground testing
  - Cost saving
  - Years before large-scale space testing available
  - Rapid turnaround for test article modification

5.3.2 Ground test environment
- Gravitational effect
  - Heat pipe capillary action
  - Concentrator structural deflection
  - Clearance of moving parts
    - Engine startup
- Atmospheric effects
  - Vacuum chamber availability
    - Full-scale concentrator
    - Full-scale radiator
      - Space temperature heat sink
  - Diffusion of solar radiation
  - Gas conduction in multi-foil insulation

Table 10.7-1. System Critical Development Issues (Continued)

- 5.3.3 Design and analysis
  - Design phase requirement
    - Feasibility of ground testing
      - Component, subsystem, system
  - Suitability of ground testing
    - Demonstrate operation and performance
  - Fixtures and procedures
- 5.4 Reliability/fault tolerance
  - 5.4.1 Essential ingredient in ranking
    - Impacts life cycle cost, weight, and area
    - Effects achievement of assigned mission
  - 5.4.2 Requirements
    - Unserviceable
      - Design to reliability
        - Component redundancy
        - Fault anticipation and analysis
      - Secondary factors
        - Cost
        - Weight and area
    - Serviceable
      - Reliability optimization
        - Options of replacement and refurbishment
        - Consequences of loss of performance
          - Partial - degradation
          - Complete loss
        - Primary factors
          - Cost
          - Weight and area
  - 5.4.3 Design and analysis
    - Design phase requirement
    - Define requirements
      - Unserviceable
      - Serviceable

Table 10.7-1. System Critical Development Issues (Continued)

- 5.5 Computer software for design optimization
  - 5.5.1 Optimization criteria
    - An arbitrary choice for technology development
      - Future spacecraft requirements not presently well defined
    - Usually a compromise of conflicting criteria
    - Process repeated after each step of technology development
      - Refinement of optimization criteria
      - Improved technical input available
  - 5.5.2 State-of-the-art power systems codes
    - Software developed to support Task II
      - Creation, modification, extension of existing codes
        - At company expense
        - Proprietary codes
      - Separate codes for Brayton, Rankine, and Stirling
        - Different level of detail
      - Common data base
        - Needs updating
          - Task III design results
          - Space Station design changes
          - A/D contract results
    - Software deficiencies
      - Alternate designs for components
        - Concentrator
        - Radiator
        - Stirling heat pipe receiver/TES/engine
        - EM pumps (ALIP and TEM)
      - Provision for excess heat rejection
      - Stirling engine characterization
        - Over a range of design conditions
      - Reliability assessment of design
    - NASA-developed software
      - Not in complete agreement with Task II software

Table 10.7-1. System Critical Development Issues (Continued)

5.5.3 State-of-the-art subsystem codes

- Number of A/D contracts in support of Space Station
- Previously developed A/D code may be proprietary and unavailable
- A/D contracts results available to power system code developers
  - Simplified algorithms for power system codes
    - Detailed subsystems codes not needed for power system

5.5.4 Assessment of code development philosophy

- Power system codes shall continue to be required
- NASA needs periodic ranking updates
  - Evaluation and redirection of A/D contracts
- NASA versus outside contractor
  - Perform ranking updates
  - Code maintenance and upgrading
  - Corresponding to technology advancements
- Formalization of power system code development
  - Develop under outside contract
    - Industrial feedback to NASA
  - More NASA involvement in code development
    - Prepare code to statement of work
    - Briefing and approval cycle
    - Deliver code to NASA
  - Documentation
  - Code update
    - Periodically over several years
    - Would require an ongoing contract
    - Close working relationship to NASA
    - Dynamic nature of code and documentation

Table 10.7-1. System Critical Development Issues (Concluded)

5.5.5 Computer software advancement

- Formalize development of power systems codes
  - Ranking of alternative power system designs
- Advance Stirling engine code development
  - Performance prediction at other design conditions
  - Off-design operation performance predictions

to verify the true worth of the anticipated advancement. Several critical development areas have been identified to be dealt with at the system level, which are: excess energy management, fabricability, ground testing, reliability/fault tolerance, and continued development of computer software for design optimization.

### 10.7.1 Excess Energy Management

Solar energy availability to a power system varies a small amount due to solar intensity, and to a larger degree due to variation in eclipse interval. For example, the two missions chosen for this study result in the following variation in energy availability:

| | Orbit Altitude, km | Inclination | Available Energy Ratio |
|---|---|---|---|
| LEO | 500 | 28.5° | 1.220(1) |
| MEO | 1200 | 0-90° | 1.573(2) |

Notes:
1. Any orbital altitude variation for LEO above 500 km will cause this value to increase somewhat.
2. For higher inclination orbits having some orbits with no eclipse.

The excess energy can be either avoided (by facing the concentrator away from the sun, by moving concentrator segments out of focus, or by shading a portion of the concentrator), rejected from the receiver/TES before the engine, bypassed around the engine to the waste heat radiator, or processed through the engine. The latter approach is the conventional choice resulting from past studies on the basis of weight and reliability concerns, but this depends on the amount of excess energy to be managed. For large amounts, one of the other choices could prove to be superior in weight and/or reliability.

To the present, solar dynamic power systems have been designed to be mission-specific. It would be desirable to conduct a study, which could be an extension of this study, regarding the design impact resulting from application of the same solar dynamic power system to LEO, MEO (mid-earth orbit), and GEO missions. Further, the study should include intermittent duty cycle (as might be required for MEO radar), the elliptical 12-hour Molnyia orbit, and possibly an Orbit Transfer Vehicle (OTV) mission which would have continuously varying solar exposure and which could benefit from intermittent duty. (Note: The OTV mission would require a much lighter weight solar power system than offered by near-term SOA hardware, whether solar dynamic or photovoltaic.)

The objective of such a study would be to determine to what degree solar dynamic power system designs can be approached in a generic fashion versus having the designs tailored to each mission. A major concern would be excess energy management, such that the optimization of management approach would need reexamination for these different mission applications.

### 10.7.2 Fabricability

This area relates more to subsystems, but it is a concern which must be introduced early in the design phase of a project. The Stirling engine has for years been a challenge to fabricability, where the end item can be properly inspected, x-rayed, and otherwise be proven sound. The area of greatest challenge for the solar dynamic power system designs of this study would appear to be the heat pipe Stirling design. The following must be accomplished, not necessarily in the order presented:

- Construction and charging of both the primary and secondary heat pipes
- Assembly of the two heat pipes to form the TES cannister and filling of the cannister
- Joining perhaps 40 of these assemblies to the Stirling engine heater head

Subsystem design and development is often divided among various contractors, which requires that interface conditions be clearly defined. For the example cited above, the issue of fabricability could be a major issue.

### 10.7.3 Ground Testing

It will be a number of years before experiments may be flown on Space Station and the competition for Shuttle space will limit that test resource. In either case, testing costs will be substantial. For reasons of time, availability and cost, there shall always be a desire and need for ground testing. Although a component or system may be destined for space application only, ground testing effects must be included in the design if such testing is to be performed. Some of the concerns to be faced in ground testing are noted in table 10.7-1.

### 10.7.4 Reliability/Fault Tolerance

This issue must be brought into the design of a system from the start, as part of the system requirements. The requirements of the customer must be realistic; in the case of serviceable spacecraft, this could be minimizing life cycle cost, whereas for an unserviceable spacecraft a certain reliability must be designed in by way of redundancy, etc.

In a sense reliability is more important than issues of weight and cost; however, achievement of a required level of reliability would ordinarily be directly reflected in weight and cost. Since weight lifting capability to orbit and financial resources are both limited quantities at any point in time, the interrelated concerns of reliability, weight, and cost may be subject to periodic review and renegotiation.

### 10.7.5 Computer Software for Design Optimization

Solar dynamic space power generation systems are being investigated for more and more applications, both civil and military. There appears to be a need for computer codes designed to make realistic estimates of system and component sizes, weights, and performance for various time frames. Mission planners need these estimates, and the emerging solar dynamic power industry needs to agree on these estimates so as to promote both the credibility and saleability of solar dynamic power.

Such a standard code does not exist. Codes in use at NASA-LeRC and the codes used in this study do not always produce results which agree, and this is to be expected. The NASA codes project what future technology may be expected to achieve, and in some instances, the timeframe and certainty of achieving the projected technology is quite indefinite. The code used in this study relied on near-term SOA, hardly considering any hardware not already in the early development phase.

Creation of a standard code would appear to be in the public interest. The code results would be no better than the data input assumptions; therefore, creation of a standard code would include:

1. A software code for sizing solar dynamic power systems for diverse power applications within certain time frames in the future
2. A data base for near-term, medium-term, and far-term SOA predictions of component performance
3. Available for use by government agencies and private industry
4. Provision for continued updating of the data base (and code) based on technological advancement

The proposed code development would be by private industry under contract to NASA-LeRC (or multi-agency grouping). Collaboration and concurrence would be of paramount importance so as to create a tool which would be readily acceptable to both private and public users. The data base and code would require periodic updating to keep pace with technological advancement.

A code such as described herein could be the centerpiece in guiding and promoting the NASA ASD Program. Investigations of the payoff for technology advances would be well founded, as the worth of the advance would be measured on the basis of being a payoff to the system as a whole. Disparities between the public and private sectors regarding the projections of power system size and performance, which today exist in the published literature, could be diminished. The customers, the mission planners, need the reliable input possible with the standard solar dynamic power system code.

## 11.0 ADVANCED TECHNOLOGY RECOMMENDATIONS FOR SOLAR DYNAMIC POWER SYSTEMS

The work reported in this section corresponds to Task V from the SOW.

### 11.1 OBJECTIVE

A recommended advanced technology program for the Brayton, Rankine and Stirling engine electric power systems shall be defined. This program shall include, but need not be limited to, the following:

1. Component materials (thermal storage, structural, coatings, insulation, etc.)
2. Component development for each power cycle
3. Component performance and endurance testing
4. System integration
5. System performance and endurance testing
    a. Ground tests
    b. Flight tests
        1) Shuttle experiments
        2) Space station dynamic system experiments
6. Control system development (sensors, actuators, etc.)
7. Identify and experimentally investigate new innovative ideas that might lead to improved system performance

### 11.2 IDENTIFICATION OF ADVANCED TECHNOLOGY PROGRAM RECOMMENDATIONS

The advanced technology program (ATP) recommendations were chosen to address the major critical development areas identified in section 10; the ATP recommendations were divided into the same groupings:

1. Concentrator
2. Receiver/TES
3. PCU
4. Radiator
5. System

The technology task descriptions are relatively brief, stating: objective, SOA, SOW, benefits, impact if technology is not developed, and technical risk. The numbers in parenthesis in the title refer back to the numbered items in the following listed tables of critical development issues from section 10:

Tables from Section 10

| | |
|---|---|
| 10.3-1 | Concentrator |
| 10.4-1 | Receiver/TES |
| 10.5-1 | PCU |
| 10.6-1 | Radiator |
| 10.7-1 | System |

There are a total of 28 ATP tasks recommended, as listed in table 11.2-1. The interrelationship of the tasks is such that many would be combined into various groupings for purposes of program execution. The tasks are divided between the subsystems as shown in figure 11.2-1. The greatest number of tasks pertain to the receiver, and the fewest to the radiator.

The need for space power systems results from the evolution of future space missions and power requirements therefore. Technology development and advanced development (see table 8.2-2) are frequently fostered by anticipated needs of future missions. The planning of technology tasks must therefore survey future expected missions within a particular time frame or series of time frames. These time frames constitute cut-off points for the entry of the various advanced development and technology development items into the design and analysis process of future power systems.

Work statements for the following technology tasks will refer to surveys of future missions. The appropriate time frame(s) must therefore be defined for each instance: fairly nearby (e.g., 1992 or 1996) for advanced development items being considered for adoption in the next generation of power systems, versus perhaps the year 2010 (as was done for this study) for those items which appear to have promise but shall require longer term technology development support.

Table 11.2-1. Recommended Advanced Technology Tasks

## 1.0 Concentrator

1-1 Ranking Concentrator Conceptual Designs
1-2 Concentrator Deployment Design Study
1-3 Concentrator Losses
1-4 Concentrator Performance Degradation
1-5 Concentrator Design, Fabrication, and Test

## 2.0 Receiver

2-1 Heat Pipe Material Selection
2-2 Individual Heat Pipe Performance Testing of Primary and Secondary Heat Pipes
2-3 Test Full-Scale Heat Pipe Receiver
2-4 TES Cannister Freeze/Thaw Cycle
2-5 TES Material Selection
2-6 TES Containment Material Compatibility
2-7 TES Material Purity
2-8 TES Containment Joining
2-9 TES Containment Filling
2-10 TES Conductance Enhancement
2-11 Aperture Shield Material Testing
2-12 Receiver Shell Material Testing

Table 11.2-1. Recommended Advanced Technology Tasks (Concluded)

## 3.0 Power Conversion Unit

3-1 Stirling Engine Design and Analysis
3-2 Design of Stirling Engine Heat Pipe Heater
3-3 High-Temperature Stirling Engine
3-4 Stirling Engine Fabrication and Test
3-5 Stirling Engine Control Design and Testing
3-6 Correlation of Stirling Engine Codes
3-7 High-Temperature Brayton Engine

## 4.0 Radiator

4-1 Radiator Heat Pipe Selection

## 5.0 System

5-1 Excess Heat Rejection
5-2 Computer Software Advancement

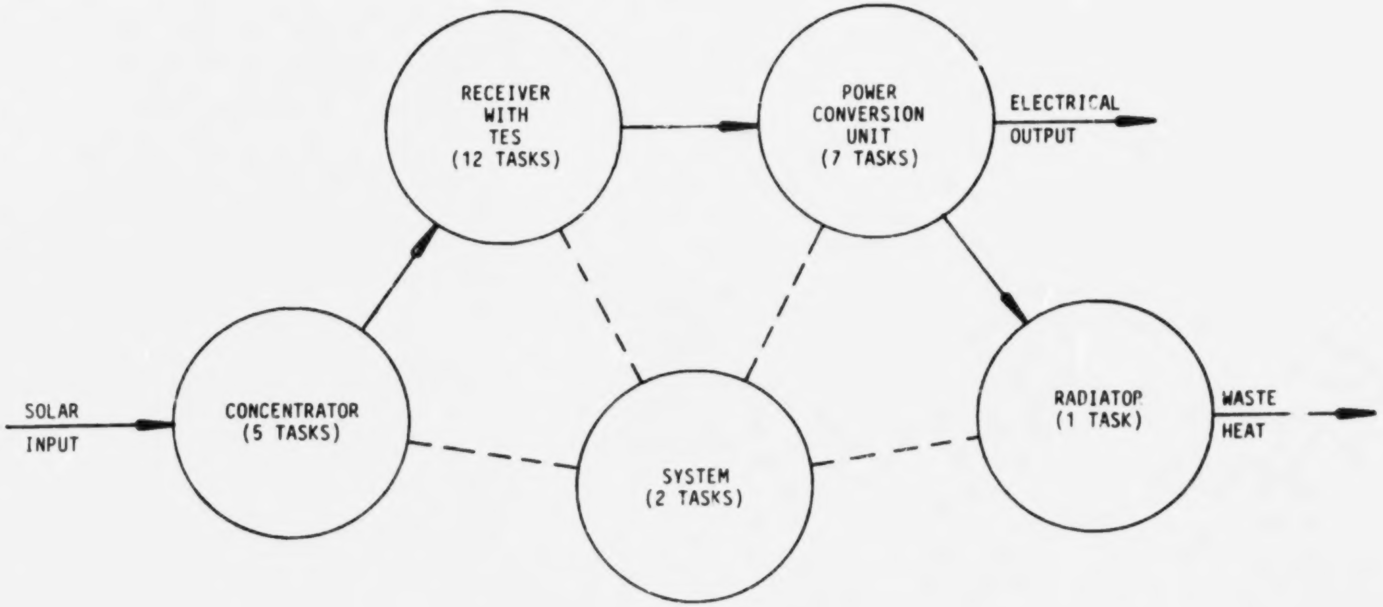

SOLAR
INPUT
CONCENTRATOR
(5 TASKS)
RECEIVER
WITH
TES
(12 TASKS)
POWER
CONVERSION
UNIT
(7 TASKS)
ELECTRICAL
OUTPUT
SYSTEM
(2 TASKS)
RADIATOR
(1 TASK)
WASTE
HEAT

Figure 11.2-1. Solar Dynamic Power System Advanced Technology Development Tasks

## 11.3 CONCENTRATOR ADVANCED TECHNOLOGY TASKS

A total of five ATP tasks were identified for the concentrator, as shown in figure 11.3-1. Descriptions of the tasks may be found at the end of section 11.3.

Two promising concentrator concepts have been investigated under NASA contracts and more activities are being planned in this area. The SRP figures presented indicate a 60% weight saving and 70% stowed volume savings in comparison to the baseline truss hex concentrator. The domed Fresnel lens concept shows similar savings may be possible. Both concepts may be automatically self-deployed and manually (or possibly automatically) restowed for retrieval. The truss hex concentrator is quite suitable for Space Station, and is both less susceptible and more easily repairable as regards micrometeoroid damage. The planned NASA activities will be targeting even lighter weight concepts.

### 11.3.1 Ranking Concentrator Conceptual Designs (Task 1-1)

This task is intended to define the drivers in concentrator design selection. Section 10.3.2 indicated that dramatic reductions in concentrator weight and stowed volume are possible. NASA-LeRC activities seek even further improvement. The ranking process is very mission dependent.

### 11.3.2 Concentrator Deployment Design Study (Task 1-2)

This task is intended to define various mission needs regarding concentrator deployment and those missions which require restow capability. Advantages and penalties associated with differing design concepts are necessary to perform comprehensive system trade studies.

### 11.3.3 Concentrator Losses (Task 1-3)

This task is intended to complement Task 1-1. Loss characteristics of each concentrator type must be examined and the payoff and penalty assessed from a power system standpoint as regards reduction of concentrator losses.

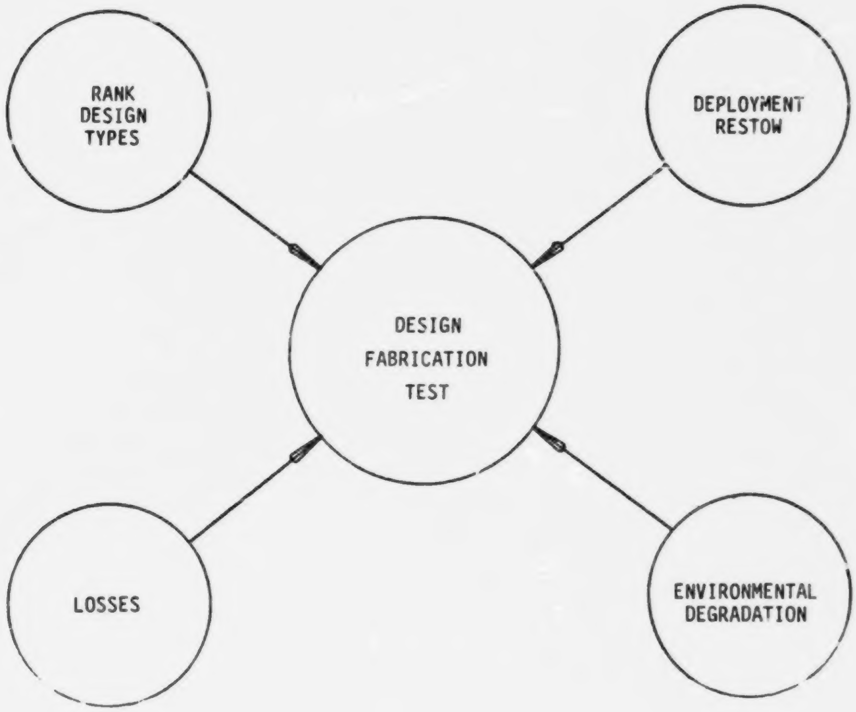

RANK
DESIGN
TYPES
DEPLOYMENT
RESTOW
DESIGN
FABRICATION
TEST
LOSSES
ENVIRONMENTAL
DEGRADATION

Figure 11.3-1. Concentrator Technology Tasks

Larger losses will be tolerable if weight and/or volume reductions are quite substantial.

### 11.3.4 Concentrator Performance Degradation (Task 1-4)

Performance degradation is primarily a materials issue. Much activity has been directed at the Space Station LEO altitudes, particularly in regard to atomic oxygen effects. Unmanned satellites will no doubt be at higher altitudes, which present different materials concerns. Degradation mechanisms vary with materials of construction and causative effect, such that some materials found unsuitable for LEO will be satisfactory at higher altitudes. Present activities in this area must be reviewed in that light and expanded as appropriate.

### 11.3.5 Concentrator Design, Fabrication, and Test (Task 1-5)

This is a customary and necessary step in the development process, providing proof of the anticipated gains and/or indication of development work yet to be accomplished.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 1-1

1. TECHNOLOGY TASK TITLE: Ranking Concentrator Conceptual Designs (1.2.3 and 1.3.3)

2. MAJOR TECHNOLOGY GROUP TITLE: Concentrator

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Concentrator
   Receiver

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Development of design concepts from the perspective of reducing concentrator weight and/or launch volume. Different missions will dictate different needs on the part of the concentrator and, therefore, different solutions and ranking criteria.

5. STATE-OF-THE-ART LEVEL: A/D work done by Harris indicates the potential for concentrators based on antenna technology to have lower weight and smaller launch volume than for the truss hex concentrator design; however, technology development is further behind and issues have yet to be worked out.

6. BRIEF WORK STATEMENT: This particular task is closely related to the design, fabrication, and test (Task 1-5); however, focuses on the two main drivers in concentrator design and optimization. Future missions will be surveyed to evaluate the limiting design criteria which should prevail, whether weight, volume, or other limitations. Existing and conceptual designs will be re-examined to reduce weight and/or volume, and at what expense in performance and cost. Any such improvements will then be examined and ranked from a power system standpoint.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 1-1
## (Concluded)

7. BENEFITS: The benefits will be establishment of rational criteria to guide further concentrator advanced development work so that efforts may be focused in the proper areas, and that improvements are properly rated as to power system benefit. Potential concentrator weight savings of 60% or more are indicated.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Ill-defined criteria by which to rank potential design improvements create the possibility of expending A/D efforts in areas not of the greatest importance.

9. TECHNICAL RISKS: A dearth of missions for advanced solar dynamic power may make this and other mission-related tasks difficult. A study is presently being conducted to evaluate differing types of electric power systems as applied to all future civilian space missions to the year 2040 (Space Station Evolutionary Power Technology Study, Contract NAS3 24902, NASA-LeRC to Rocketdyne, September 1986 to present). The study should identify a number of applications for solar dynamic power.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 1-2

1. TECHNOLOGY TASK TITLE: Concentrator Deployment Design Study (1.4.3)

2. MAJOR TECHNOLOGY GROUP TITLE: Concentrator

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Concentrator

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Determination of typical requirements for concentrator deployment and restow, and to expand upon such work already performed so as to determine the advantages and penalties associated with differing requirements.

5. STATE-OF-THE-ART LEVEL: Some attention has been given to deployment and restow for the A/D work by Harris. Harris is also working out the deployment for the Space Station truss hex design.

6. BRIEF WORK STATEMENT: Future missions will be surveyed to determine deployment requirements; whether man-assisted or not, the need for restow, the need for multiple deployments, etc. Existing and conceptual designs will be re-examined with regard to the different deployment and restow requirements to determine impact on weight, launch volume, cost, and performance.

7. BENEFITS: Determination of impact on weight, launch volume, etc. will provide useful information for spacecraft designers as to the advantages and penalties associated with the alternative deployment requirements which the designers might be considering.

# SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 1-2
(Concluded)

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Inadequate definition of future spacecraft requirements creates the possibility of expending A/D efforts in areas not of the greatest importance. Misconceptions of solar dynamic power system capabilities, or lack of valid information could cause solar dynamic power to be eliminated from consideration.

9. TECHNICAL RISKS: The design requirements for deployment will depend on the quality and quantity of missions available for survey, and that availability may be quite limited, as regards deployment information.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 1-3

1. TECHNOLOGY TASK TITLE: Concentrator Losses (1.6.4)

2. MAJOR TECHNOLOGY GROUP TITLE: Concentrator

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Concentrator
   Receiver

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Evaluation of concentrator losses and recommendation of a development program to reduce these losses insofar as power system performance is improved. Identify typical spacecraft interactive effects, recognizing that the solar power system is an appendage to the spacecraft, and that other spacecraft will interact with the power system much differently than will Space Station.

5. STATE-OF-THE-ART LEVEL: Solar tracking has thus far been earth-bound for solar concentrators, as all space power has been generated using photovoltaic arrays. Few large concentrators have been built, and each design has its own inherent errors. Except for antenna, no large space structure has attempted the error limitation required for solar dynamic power.

6. BRIEF WORK STATEMENT: Conduct an analysis of concentrator losses (slope error, non-specularity, pointing error, and segmented surface effects) and causes. Establish state-of-the art capability in minimizing the causative effects. Perform trade studies as to improvement in power system performance (weight) through error reduction, and determine the probable means of error reduction. The trade study will establish error reduction goals related to the point of diminishing returns. A recommended development plan will be prepared establishing improvement goals for future missions, for different concentrator configurations, and establishing component and subassembly tests required.

### SOLAR DYNAMIC POWER SYSTEM

### TECHNOLOGY TASK 1-3

(Concluded)

7. BENEFITS: The several loss mechanisms will be examined collectively, including interactions between concentrator and receiver. The measure of improvement will be power system performance, thus keeping the effect of the losses in perspective. No remarkable reduction in losses may be expected; rather, maintaining loss levels while dramatically reducing concentrator weight and/or cost would be a major objective.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Without a proper analysis of losses, what may be achieved presently or in the future will not be known with any certainty, as little more than present Space Station studies exist, and there is no demonstration work to back up the studies. Inadequate definition of the system payoff due to reduction of losses creates the possibility of expanding A/D efforts in areas not of the greatest importance.

9. TECHNICAL RISKS: Risk is minimized by the conduct of analytical trade studies, with effects carried to the power system level, before a recommendation is made as to a development program.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 1-4

1. TECHNOLOGY TASK TITLE: Concentrator Performance Degradation (1.7.4)

2. MAJOR TECHNOLOGY GROUP TITLE: Concentrator

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Concentrator
   Balance of power system sizing

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: To assess degradation potential (primarily optical performance) for the concentrator for typical lifetimes, with or without periodic upgrading, and to assess the impact on other power system components. The impact may be in oversizing or thermally overloading components, for which there must be appropriate accommodation.

5. STATE-OF-THE-ART LEVEL: No long-term flight experience is available for concentrator materials; what is available comes from brief flight experiments and laboratory experiments simulating the space environment. Space Station work is concentrating on manned or man-tended spacecraft located at LEO which can be periodically refurbished, upgraded, or replaced.

6. BRIEF WORK STATEMENT: Conduct an analysis of concentrator degradation mechanisms for a variety of future missions. Analyze the degradation estimated to occur for typical spacecraft lifetimes. Assess penalties to the power system to accommodate the degradation (weight, area, volume, excess thermal management, etc.). Examine the feasibility of periodic replacement or refurbishment. A recommended development plan will be prepared establishing improvement goals for future missions, for different concentrator configurations (i.e., reflective versus refractive), and establishing component and subassembly tests required.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 1-4
## (Concluded)

7. BENEFITS: Improved knowledge of concentrator degradation permits more accurate sizing of the concentrator and other power system components, and permits a better assessment of the feasibility of periodic refurbishment or replacement of the concentrator.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Uncertainties regarding degree of degradation to expect will lead to improper sizing of the concentrator and other power system components. An improperly sized power system would result in a penalty of being either overweight or underpowered.

9. TECHNICAL RISKS: Risk is minimized by the conduct of analytical trade studies, with effects carried to the power system level, before a recommendation is made as to a development program.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 1-5

1. TECHNOLOGY TASK TITLE: Concentrator Design, Fabrication, and Test (1.1.4 and 1.1.5)

2. MAJOR TECHNOLOGY GROUP TITLE: Concentrator

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Concentrator
   Receiver

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Further develop different concentrator designs for different missions (i.e., orbit location, limitations on size or weight) to support a 35-40 kWe dynamic power system.

5. STATE-OF-THE-ART LEVEL: The truss hex structure is being developed for Space Station. A/D contract work by Harris and has produced conceptual designs for concentrators based on antenna technology.

6. BRIEF WORK STATEMENT: Conduct an analysis of future mission needs and limitations, and the application of various concentrator design concepts to those missions. Perform design tradeoffs and rank the conceptual designs. Perform detailed designs for one or more configurations. Establish need and objectives in regard to fabrication and test of these designs in order to advance the technology of each design, to where objective choices may be made for future mission applications. Design tradeoffs regarding support structure, surface design, etc. will be considered for breakout as separate A/D tasks to support this task.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 1-5
(Concluded)

7. BENEFITS: The benefits sought will be improved power system performance resulting from concentrator improvements, whether improved thermal performance, reduced weight, reduced launch volume, etc. Of particular importance will be weight reduction.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): The concentrator constitutes a significant portion of power system weight and launch volume; thus, an absence of development work here would limit performance to that which is developed for the Space Station.

9. TECHNICAL RISKS: Risk is minimized by expanding upon ongoing concentrator A/D contracts, reducing conceptual designs to detailed designs, fabrication, and test of one and preferably more design configurations.

## 11.4 RECEIVER/TES ADVANCED TECHNOLOGY TASKS

A total of twelve ATP tasks were identified for the receiver/TES, as shown in figure 11.4-1. Descriptions of the tasks may be found at the end of section 11.4.

The heat pipe Stirling power system conceptual design developed under this study indicates the potential of over 20% weight savings as compared to the CBC configuration. The proposed heat pipe receiver/TES design is an integral part of the Stirling cycle design, and that design must be carried into the development stage as is being done for the Boeing A/D CBC receiver/TES. The CBC receiver intervals used in this study duplicate the A/D receiver design, scaled for power.

Conductance enhancement for CBC is provided by inclusion of 20% dense nickel felt within the TES space. The same approach using 16% dense nickel felt is proposed for the heat pipe Stirling design. The nickel felt would equal LiF weight for 16.6% density, and containment volume is increased by presence of the felt. section 10.4.2 recommends consideration of alternate felts, up to and including graphite fiber felt with the potential to increase felt conductivity by 7 times and to reduce felt weight (material density) by 4 times for the graphite. Compatibility of the alternate felt and the TES material is not known. There is a substantial potential for improvement in any receiver/TES design as a result of increased TES material section thickness, reduction in felt weight and TES containment weight, and the benefits of reduced thermal stress and improved engine operation. Estimates of the possible weight savings have not been made; however, for the heat pipe Stirling design, the ratio of containment plus felt metal weight to LiF weight is 3.16:1 (including receiver heat pipe weights). Reducing this ratio by one-third or more appears to be a reasonable goal, especially if graphite felt could be used. As a backup to the graphite fiber, nickel coated graphite fibers should work out quite well, improving fiber conductivity perhaps 4-5 times over nickel fiber.

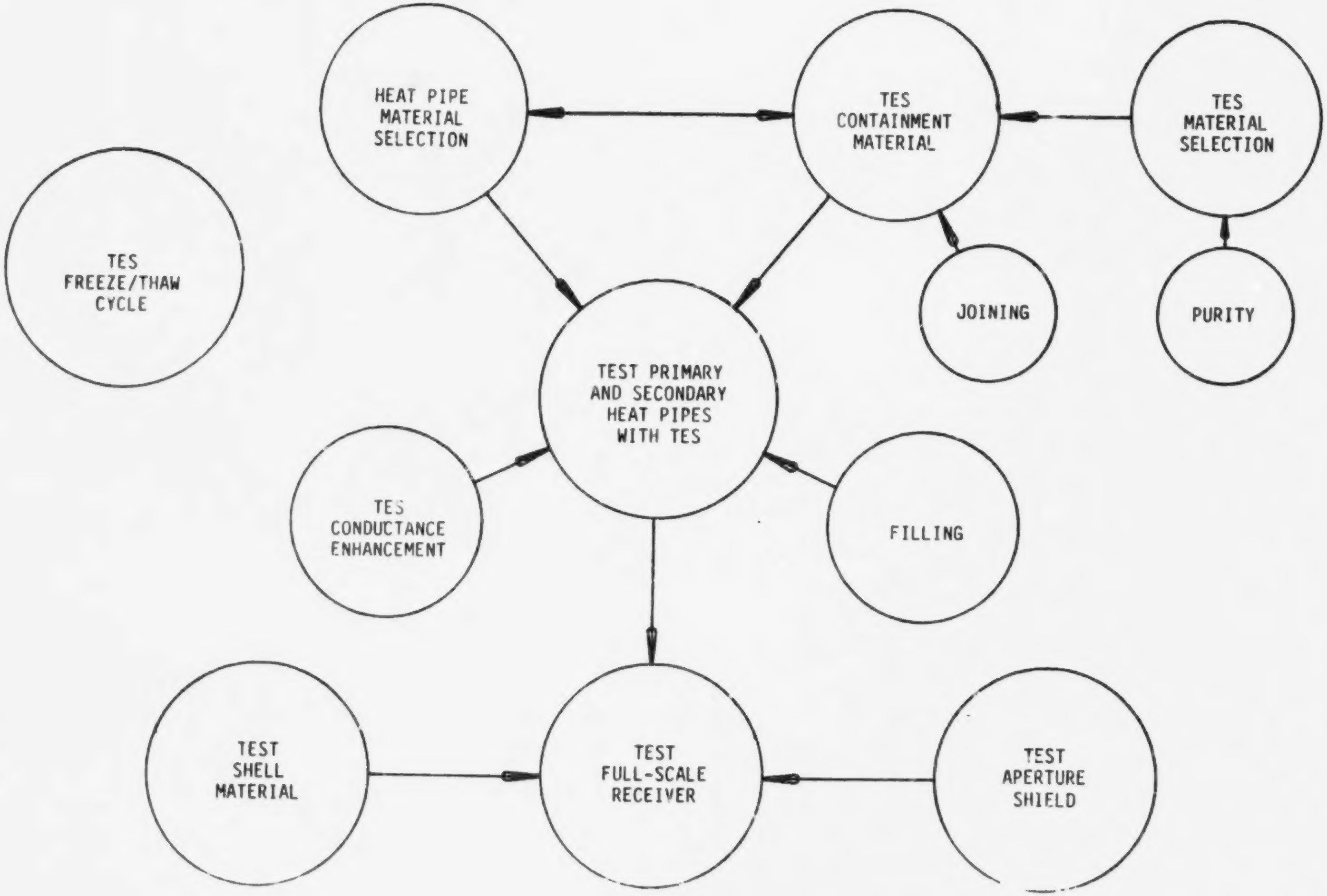

HEAT PIPE MATERIAL SELECTION
TES CONTAINMENT MATERIAL
TES MATERIAL SELECTION
TES FREEZE/THAW CYCLE
JOINING
PURITY
TEST PRIMARY AND SECONDARY HEAT PIPES WITH TES
TES CONDUCTANCE ENHANCEMENT
FILLING
TEST SHELL MATERIAL
TEST FULL-SCALE RECEIVER
TEST APERTURE SHIELD

Figure 11.4-1. Receiver/TES Technology Tasks

The recommended ATP tasks pursue a variety of related tasks so as to verify compatibility, TES filling, etc., and to further develop the pumped loop Stirling receiver as a backup to the heat pipe receiver. Tasks are also recommended relating to receiver shell and insulation for TES, and aperture shield.

### 11.4.1 Heat Pipe Material Selection (Task 2-1)

The first three tasks (2-1 to 2-3) are designed to lead to construction and test of a full-scale Stirling heat pipe receiver/TES with simulated solar input and simulated engine thermal load. This task also relates directly to the TES containment material task (2-6), as the heat pipes and TES share common walls. The Stirling heat pipe receiver/TES design developed for this study is a conceptual design, unlike the CBC designs of reference 1 and the Boeing A/D design. The heat pipe design will require analysis of both TES material compatibility and operating temperature and selection of a refractory material may be necessary for the heat pipes. If that is the case, a transition joint to a superalloy material would be required before connection to the engine heaters. This task includes a materials test program.

### 11.4.2 Individual Heat Pipe Performance Testing of Primary and Secondary Heat Pipes (Task 2-2)

This task recommends construction and test of the primary and secondary heat pipes as individual units. Upon successful completion of the tests, one or more complete heat pipe/TES units would be assembled and tested. The object of the unit testing would be development and proof of operation and performance prior to full scale receiver testing.

### 11.4.3 Test Full-Scale Heat Pipe Receiver (Task 2-3)

A full-scale ground test receiver/TES assembly would be tested in a vacuum chamber with both simulated solar input and engine thermal load. This testing would provide proof of operation and/or indication of development work yet to be accomplished.

### 11.4.4 TES Cannister Freeze/Thaw Cycle (Task 2-4)

This technology task is directed at the pumped loop receiver/TES design concept presented in this study. The purpose is to investigate the perceived cannister freezing problem (as described in section 10.4.3), to proposed design alternatives for the pumped loop TES, and to performance analysis and laboratory testing of viable alternatives. The pumped loop concept is a backup to the heat pipe concept; however, no full scale testing is recommended.

### 11.4.5 TES Material Selection (Task 2-5)

This study has established that LiF salt is a superior TES material from a power system performance standpoint (minimum weight). Continuation of study in the area of TES material selection is recommended. Component design advancement could possibly alter conclusions of this study; however, dramatic changes in receiver/TES design would be necessary to alter this conclusion. More probably, any change would occur as a result of discovery of an alternative TES material. This task would continue present NASA-LeRC activities in this area, both in-house and those being conducted at Oak Ridge National Laboratory.

### 11.4.6 TES Containment Material Compatibility (Task 2-6)

This task is primarily a materials test program. Much activity is presently going on in this area in support of the Space Station CBC power system, designed to use an eutectic mixture of $LiF-CaF_2$ which melts at 79K (142F) lower temperature than LiF. In addition, NASA-LeRC is conducting an extensive in-house program in this area. This task would continue and expand that work as necessary, in particular tying into Task 2-1, the heat pipe materials selection task. A key issue in these several tasks will be determination if the superalloys will be adequate for the case of LiF TES, or if refractory alloys will be required. Alternate TES materials (with different melting temperatures) as might emerge from Task 2-5 results would impact the containment material activity.

### 11.4.7 TES Material Purity (Task 2-7)

This task is an essential counterpart of the previous task on containment material compatibility (Task 2-6) and is noted separately to emphasize its importance. Frequently, with the containment of chemically reactive materials, it is the presence of impurities that initiates the corrosion process. A frequent practice is to include a small amount of material, such as Li metal in the LiF, which would tend to be more reactive and thus, neutralize the impurities. In the final system, the source of the impurities may be the TES material proper, or may be present in the TES containment at time of fill, or may be picked up in the fill process. This task interrelates with numerous other tasks in the process of producing a properly filled TES container.

### 11.4.8 TES Containment Joining (Task 2-8)

The TES containment will involve built-up assemblies with welded joints. There will also be closure welds as a result of the TES filling process. These welds must be chemically and mechanically compatible with the TES material to avoid problems of weld-induced failure. (Mechanical compatibility relates to crevices or stress risers which would promote corrosion whereas chemical compatibility refers to the weld grain composition as compared to the parent material.) This task would interrelate with Tasks 2-6 and 2-7.

### 11.4.9 TES Containment Filling (Task 2-9)

The procedures and hardware for filling the TES containers will be quite involved, based on experiences related to the Space Station CBC TES material handling. For example, issues to be resolved for the heat pipe Stirling TES configuration include: higher temperature, the presence of felt material in the TES space, the TES space being accessible from one end only for filling, the physical size (length) of the assembled unit with heat pipes, and handling for cool-down to ensure proper wicking and void location. This pilot program task would seek to examine the impact of receiver/TES related issues to the design of a TES fill facility, and to establish the techniques and procedures required to ensure a successful filling operation.

### 11.4.10 TES Conductance Enhancement (Task 2-10)

As was discussed in section 10.4.2, TES conductance enhancement is quite important to receiver/TES design as TES section thickness is dictated by conductivity and the required heat transport rate. Improved conductance allows thicker TES material sections, variation in containment configuration for weight reduction, can minimize temperature differences and related stresses, and reduce temperature variations to the engine. This task has the potential for substantial payoff in receiver/TES weight reduction and may substantially effect receiver/TES configuration. As such, it should be a high priority task, picking up where the Space Station A/D receiver activity leaves off. This is an analysis and test program requiring early initiation of conductance enhancement material compatibility testing (e.g. the various felts mentioned in 10.4.2, including graphite fiber felt).

### 11.4.11 Aperture Shield Material Testing (Task 2-11)

This task seeks to further study the requirements and constraints placed upon aperture shield design, and to recommend alternative design approaches. Finite life and materials contamination related to a sacrificial material design (i.e., a graphite shield) may not be tolerable for an unserviceable spacecraft, and may well require development of a design tolerant of the expected thermal flux levels. Development testing would be performed.

### 11.4.12 Receiver Shell Material Testing (Task 2-12)

This analysis and test program would examine alternative receiver shell designs with an objective of minimizing power system weight as a function of the shell weight combined with the effect of thermal losses upon the concentrator size and weight. Laboratory testing upon test wall segments would be performed as needed to confirm predicted thermal characteristics. The same/similar shell configuration would be developed for insulation of other major high temperature components, such as remote TES storage.

## SOLAR DYNAMIC POWER SYSTEM

## TECHNOLOGY TASK 2-1

1. TECHNOLOGY TASK TITLE: Heat Pipe Material Selection (2.1.3)

2. MAJOR TECHNOLOGY GROUP TITLE: Receiver, Stirling Cycle

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Receiver
   Thermal energy storage (TES)
   Stirling engine

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Determine suitable materials for the construction of high temperature heat pipes for use in a solar receiver. Temperature corresponds to LiF TES, which melts at 1121K (1558F).

5. STATE-OF-THE-ART LEVEL: Very limited experience with materials exposed to liquid metal heat pipe working fluid at elevated temperatures. Related experience primarily at Los Alamos National Laboratory (LANL).

6. BRIEF WORK STATEMENT: Perform analysis and laboratory testing to determine suitable materials for high-temperature heat pipes. The testing will be performed using coupons for initial screening, followed by testing of stressed samples. This work must be closely coordinated with the TES material compatibility work (Task 2-6), since in some receiver designs, the heat pipe material will be exposed to both the heat pipe fluid and the TES material.

7. BENEFITS: The benefits of a heat pipe solar receiver will be reduced power system weight. This will be due to higher allowable average flux levels resulting in a smaller cavity and a higher Stirling cycle efficiency (higher $T_H$) reducing size and weight of other major components, as well as the heat pipe design resulting in a lower weight receiver/TES subsystem.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 2-1
## (Concluded)

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): The alternative concept for heat transport is a pumped liquid metal loop. Power system weight is estimated to be 10-15% higher, and reliability would be lower due to use of a high-temperature EM pump.

9. TECHNICAL RISKS: Risks are minimized with a two-step approach - first, testing coupons and then testing of stressed samples of candidates passing the coupon test. Coordination of this work and the TES containment material work will permit more rapid determination of materials compatibility with both the heat pipe fluid and the TES material.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 2-2

1. TECHNOLOGY TASK TITLE: Individual Heat Pipe Performance Testing of Primary and Secondary Heat Pipes (2.1.4)

2. MAJOR TECHNOLOGY GROUP TITLE: Receiver, Stirling Cycle

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Receiver
   Thermal energy storage (TES)
   Stirling engine

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Determine heat pipe operation at nominal and above-nominal heat transfer rates. Determine the burnout limits of the primary heat pipe. Determine thermal performance characteristics of the heat pipe/TES integrated assembly.

5. STATE-OF-THE-ART LEVEL: Very limited experience in high-temperature liquid-metal heat pipes. Related experience primarily at LANL.

6. BRIEF WORK STATEMENT: Design, construct, and test full-scale heat pipes, both primary and secondary pipes tested individually. Heat pipe designs must be suitable for operation under 1-g environment. Testing may be of single or few-pipe setup. The objective is proving the thermal performance of the heat pipes and determination of the burnout limitations of the primary heat pipe design. Testing will subject the primary heat pipe to typical nonuniform axial heat flux profiles. Testing at different inclination angles will provide useful information for full-scale receiver testing.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 2-2
## (Concluded)

Upon successful completion of individual heat pipe testing, the combined primary heat pipe/TES/secondary heat pipe assembly will be assembled and tested. Testing may be of single or few-pipe setup. Testing will be with simulated solar heat flux input and engine heat load. Testing will simulate sunlight and eclipse intervals to provide testing of the TES to prove that thermal charge and discharge performance is as designed.

7. BENEFITS: The proof of satisfactory heat pipe TES design and operation will lead to lower power system weight as a result of employing a heat pipe receiver.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): The alternative pumped loop receiver design results in increased power system weight and reduced reliability.

9. TECHNICAL RISKS: Risks are minimized by testing the primary and secondary heat pipes first individually, then integrated with TES before integration with the full-scale receiver and the engine.

## SOLAR DYNAMIC POWER SYSTEM

## TECHNOLOGY TASK 2-3

1. TECHNOLOGY TASK TITLE: Test Full-Scale Heat Pipe Receiver (2.1.5)

2. MAJOR TECHNOLOGY GROUP TITLE: Receiver, Stirling Cycle

3. POWER SYSTEM COMPONENT(S) EFFECTED:

    Receiver
    Thermal energy storage (TES)
    Stirling engine

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Demonstrate full-scale heat pipe receiver operation and performance, integrating the primary heat pipes, thermal energy storage (TES), and secondary heat pipes.

5. STATE-OF-THE-ART LEVEL: Nothing like this heat pipe receiver has been developed and tested. The proposed Space Station ORC receiver employed a heat pipe design with far lower temperature and altogether different TES design approach than chosen here.

6. BRIEF WORK STATEMENT: Design, construct, and test a full-scale heat pipe receiver with simulated solar heat flux input and engine heat load. Should analysis or individual heat pipe testing indicate problems with testing the flight design heat pipes at 1-g, then alteration of the axisymmetric design or a 2-dimensional segment will be adopted for testing. Testing will be conducted in a vacuum chamber with infrared heat input. A moveable heat lamp assembly will be needed to simulate sunlight and eclipse intervals. A flight design shell may be used for testing, although that is not necessary.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 2-3
## (Concluded)

7. BENEFITS: The proof of satisfactory heat pipe receiver operation culminates stepwise development tasks, beginning at materials selection. Each step allows re-evaluation and justification of the weight and performance advantages of the heat pipe receiver over the pumped loop receiver.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): The alternative pumped loop receiver design results in increased power system weight and reduced reliability. The weight advantage of the heat pipe receiver results from reduced receiver/TES weight, plus reduced power system weight realized from higher Stirling engine efficiency.

9. TECHNICAL RISKS: Risks are minimized for receiver development with the successful performance of several technology tasks necessary before proceeding to full-scale receiver development. Backup development will be performed on the pumped loop receiver design as an alternate to the heat pipe receiver.

## SOLAR DYNAMIC POWER SYSTEM

## TECHNOLOGY TASK 2-4

1. TECHNOLOGY TASK TITLE: TES Cannister Freeze/Thaw Cycle (2.3.2)

2. MAJOR TECHNOLOGY GROUP TITLE: Receiver, Pumped Loop

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Thermal energy storage (TES)

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Demonstrate that LiF TES cannisters used for pumped loop thermal storage may be progressively frozen from one end to the other and then progressively thawed from the end first frozen, as would occur for the conceptual design chosen for pumped loop TES.

5. STATE-OF-THE-ART LEVEL: Rocketdyne has demonstrated such a TES unit using LiOH TES material, which expands only 1.5% upon melting; whereas, LiF expands ~30%.

6. BRIEF WORK STATEMENT: First, to test LiF-filled cannisters (of a size approximately 1" x 48") freezing and thawing alternately from the same end to determine whether this may be successfully done without either additives to the LiF or by compartmentalizing of the LiF. Presuming that this testing will not be successful, devise alternate designs for testing which will accommodate the freeze-thaw cycle. Additives to be examined will include addition of a small amount (<10%) of Li with the objective to form a slush, or inclusion of metallic foam (or nonmetallic) for the control of void formation. Compartmentalization (shorter cannister lengths) and alternate diameters will also be investigated. System weight optimization will be considered if more than one viable solution is found for thermal energy storage.

# SOLAR DYNAMIC POWER SYSTEM
# TECHNOLOGY TASK 2-4
(Concluded)

7. BENEFITS: The pumped loop remote thermal storage design is a straightforward design concept, quite similar to a shell and tube heat exchanger. It is relatively lighter weight, as other pumped loop design alternatives appear to entail increased TES containment weight. The pumped loop receiver is necessary as a backup to the heat pipe receiver concept.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): A proven workable TES concept is required to support the pumped loop receiver design concept. Considering the large volumetric change occurring upon melting LiF or other candidate TES materials, there is no concept presently proven. The pumped loop receiver is the backup concept for the heat pipe receiver.

9. TECHNICAL RISKS: Risks will be minimized by designing a number of alternative concepts for laboratory testing to ensure that a workable solution is found. Long-duration cyclic testing of one or more candidate configurations will be performed to prove durability.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 2-5

1. TECHNOLOGY TASK TITLE: TES Material Selection (2.4.1)

2. MAJOR TECHNOLOGY GROUP TITLE: Receiver

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Receiver
   Thermal energy storage (TES)

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Determine other viable TES materials which would reduce power system weight over the baseline LiF TES designs, whether for Brayton or Stirling power cycles.

5. STATE-OF-THE-ART LEVEL: Analytical studies have chosen LiF TES over other materials due to high heat of fusion, which results in lower power system weights than for higher temperature cycles with higher engine performance.

6. BRIEF WORK STATEMENT: Conduct analysis and test of alternate TES materials such as eutectic salt mixtures or eutectic silicon/metal mixtures. Analytically selected candidate mixtures will be laboratory tested for basic properties pertinent to use as TES (density, melting temperature, narrow melt range, heat of fusion, expansion coefficient). Promising candidate materials will be examined for power system design impact, especially weight and operating temperature.

7. BENEFITS: Benefits sought will be power system weight reduction, improved material compatibility, thermal conductivity, etc., as compared to the baseline LiF TES.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 2-5
## (Concluded)

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Based on power system weight, the next most viable TES candidate was NaF, which melts at 1261K (1810F) versus 1121K (1558F). Due to the significantly higher operating temperatures for the receiver, TES, and engine, NaF is not a backup which could be readily substituted for LiF. The present backup to LiF would be the material selected for Space Station CBC, although lower performance would result (larger area and weight, lower efficiency).

9. TECHNICAL RISKS: There are no assurances that a superior TES material may be found; however, work of this kind is necessary at some point to assure that a technically superior design for TES is achieved.

## SOLAR DYNAMIC POWER SYSTEM

## TECHNOLOGY TASK 2-6

1. TECHNOLOGY TASK TITLE: TES Containment Material Compatibility (2.5.1)

2. MAJOR TECHNOLOGY GROUP TITLE: Receiver

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Receiver
   Thermal energy storage (TES)

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Determine suitable materials for the containment of the TES material (baseline LiF). The materials must also be compatible with the external heat transfer medium, whether liquid metal or vapor.

5. STATE-OF-THE-ART LEVEL: Some laboratory testing has occurred at Rocketdyne and Boeing in support of Space Station Phase B work.

6. BRIEF WORK STATEMENT: Perform analysis and laboratory testing to determine suitable materials for high-temperature storage of selected TES materials. Baseline TES is LiF salt, and other materials may result from the TES material selection (Task 2-5). The testing will be performed on coupons for initial screening, followed by testing of stressed samples. This work must be closely coordinated with the heat pipe material selection work (Task 2-1), since the TES containment material will be externally exposed to the heat pipe liquid metal and vapor, or exposed to NaK in the case of the pumped loop receiver.

7. BENEFITS: Benefits sought will be power system weight reduction, improved material compatibility, and tolerance to TES material impurities.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 2-6
(Concluded)

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): All solar dynamic power cycles under consideration employ thermal energy storage to provide power during the eclipse part of an orbit. Therefore, suitable TES containment material is required. A backup of battery storage or fuel cells for eclipse, and shutdown of the solar dynamic unit during eclipse, is not considered a viable backup except for non-earth orbiting missions such as lunar or Mars.

9. TECHNICAL RISKS: Risks are minimized with a two-step approach - first, testing coupons and then testing of stressed samples of the candidates passing the coupon test. Coordination of this work with the heat pipe material selection work will permit more rapid determination of satisfactory materials for TES containment.

## SOLAR DYNAMIC POWER SYSTEM

## TECHNOLOGY TASK 2-7

1. TECHNOLOGY TASK TITLE: TES Material Purity (2.5.2)

2. MAJOR TECHNOLOGY GROUP TITLE: Receiver

3. POWER SYSTEM COMPONENT(S) EFFECTED:

    Thermal energy storage (TES)

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Determine the purity level required for the TES material to avoid deleterious effects upon the TES containment material. Determine how to achieve the required purity level.

5. STATE-OF-THE-ART LEVEL: Laboratory work has been directed at the compatibility of the TES material and the containment material, and the matter of impurities has been dealt with incidentally rather than as a separate task.

6. BRIEF WORK STATEMENT: This task is somewhat of a tradeoff in that use of extremely pure TES, proper container preparation, and filling procedures will tend to eliminate the effects of impurities. Alternatively, analysis and testing of the containment material tolerance to impurities is also a significant task in terms of time and money. This task will take a two-pronged approach to seek a proper solution (e.g., which approach will be preferred). The work will include analysis and laboratory verification of typical impurities found, methods of refinement, containment material tolerance to impurities, container preparation, and filling procedures.

7. BENEFITS: Establishing a good understanding of the role of impurities in TES materials will permit establishing specifications and procedures necessary for producing a long-life reliable thermal energy storage subsystem.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 2-7
## (Concluded)

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Disregard for the effect of impurities could produce premature failures, essentially due to inadequate quality control.

9. TECHNICAL RISKS: Work here must be thorough enough that long-term failure effects are uncovered and corrected so as to achieve design lifetime of the TES containment.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 2-8

1. TECHNOLOGY TASK TITLE: TES Containment Joining (2.5.3)

2. MAJOR TECHNOLOGY GROUP TITLE: Receiver

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Thermal energy storage (TES)

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Establish necessary techniques and procedures for welding of the TES containment material.

5. STATE-OF-THE-ART LEVEL: Considerable information is being generated for the Space Station dynamic power systems which are considering use of salt mixtures containing LiF for the CBC cycle TES. Garrett, Boeing, and Rocketdyne have each done some work in this area.

6. BRIEF WORK STATEMENT: Perform analysis and laboratory testing of techniques and procedures necessary to accomplish satisfactory welding of containment materials chosen for TES containment. Of necessity, this work must be coordinated with TES containment material selection (Task 2-6) and TES material purity (Task 2-7).

7. BENEFITS: Proper weld qualification procedures are necessary for producing a long-life reliable thermal energy storage subsystem.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Improper weld procedures could result in premature TES failures, and result in compromise or failure of the power system.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 2-B
## (Concluded)

9. TECHNICAL RISKS: Work here must be thorough enough that long-term failure effects are uncovered and corrected so as to achieve design lifetime of the TES containment.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 2-9

1. TECHNOLOGY TASK TITLE: TES Containment Filling (2.5.4)

2. MAJOR TECHNOLOGY GROUP TITLE: Receiver

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Thermal energy storage (TES)

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Establish necessary techniques and procedures for filling the TES containers.

5. STATE-OF-THE-ART LEVEL: Some experience has been gained in support of the Space Station CBC system, which has proposed use of a eutectic mixture including LiF. Little experience has occurred for pure LiF. Boeing and Rocketdyne have performed recent experimental work in support of Space Station.

   The Boeing A/D receiver configuration is similar to the Stirling heat pipe receiver in that each are designed with a low number of TES containment volumes (24 and 40 respectively, see figures 6.2.1.2-2 and 2.4-2), and utilize felt metal within the containment volume. In comparison, the Space Station CBC receiver design employs 82 working fluid tubes, each fitted with 96 TES canisters, for a total of 7872 canisters. Containment volumes with felt metal will have to be filled with molten salt, whereas the small canisters for the Space Station CBC receiver design may be filled with a measured quantity of solid material.

6. BRIEF WORK STATEMENT: Perform analysis and laboratory testing of techniques and procedures necessary to successfully fill TES storage containers with LiF. Many containment geometries are possible; the configurations of annular containment for CBC and heat pipe Stirling, and the cannister configuration for the pumped loop Stirling, will be

considered along with other designs which may emerge from advanced concept receiver studies presently being studied by Garrett and Sanders. It will be essential to coordinate this work with the conductance enhancement work (Task 2-10) and the cannister freeze/thaw work (Task 2-4).

7. BENEFITS: Qualified procedures are necessary in order to reliably accomplish the task of TES containment filling. Should the results of this task cause particular features to be incorporated into the containment design, then it is important that information be available to effect receiver subsystem design, performance, and weight.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Improper procedures could produce TES storage with insufficient thermal storage. Delays in establishing qualified procedures could result in possible redesign efforts and delays in accomplishing receiver subsystem test programs.

9. TECHNICAL RISKS: Technical risks are minimized by addressing TES containment filling as a separate task early in power system development work.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 2-10

1. TECHNOLOGY TASK TITLE: TES Conductance Enhancement (2.6.2)

2. MAJOR TECHNOLOGY GROUP TITLE: Receiver

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Thermal energy storage (TES)

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Demonstrate methods of improving heat transfer into and out of the TES material with the objectives of (1) reducing temperature variations at engine inlet, and (2) reducing overall power system weight (whether the TES subsystem weight is decreased or increased).

5. STATE-OF-THE-ART LEVEL: Preliminary analysis and laboratory testing have been conducted by Boeing, Sundstrand, Rocketdyne, and others.

6. BRIEF WORK STATEMENT: Perform an analysis and test program to determine conductance enhancement benefits and penalties for several possible techniques, including mechanical enhancement, additives, and/or alternative containment geometries. Various techniques and geometries will be screened using finite element thermal models, and the results compared to the case of no conductance enhancement. The more promising techniques will be compared on a power system weight basis, accounting for both TES subsystem weight changes and any engine performance improvement effects. A test program based on the analytical results will examine the more promising conductance enhancement techniques, including at least one mechanical enhancement approach, and also the case of Li additive to LiF (to support the pumped loop receiver design). Testing results will be correlated with output of the thermal models and the analytical screening results updated as necessary.

## SOLAR DYNAMIC POWER SYSTEM

## TECHNOLOGY TASK 2-10

(Concluded)

7. BENEFITS: The benefits sought are a reduction in power system weight and reduction in temperature variation at the engine inlet. The latter benefit in itself is expected to contribute to reduced power system weight due to improved engine performance and secondly, to possibly simplify engine control.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): The probability is a heavier power system weight and necessity of coping with wider variations in engine inlet temperature.

9. TECHNICAL RISKS: Risks are minimized by performing extensive analysis of various enhancement techniques to narrow the number of choices to be tested. Correlation of test results and the analytical models provides confidence in the final recommendations regarding conductance enhancement.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 2-11

1. TECHNOLOGY TASK TITLE: Aperture Shield Material Testing (2.7.3)

2. MAJOR TECHNOLOGY GROUP TITLE: Receiver

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Receiver

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Determine suitable material and design for a receiver aperture shield capable of adequate protection for a concentration ratio of 2000 or more.

5. STATE-OF-THE-ART LEVEL: The higher cycle operating temperatures will require higher concentration ratios (2000 or more) than currently proposed for Space Station CBC (1100-1200). Aperture shield designs by Garrett and Boeing are substantially different, possibly due to differing assumptions, and applicable testing has not been performed.

6. BRIEF WORK STATEMENT: Design requirements for anticipated operating scenarios will be defined as regards concentration ratio, time of exposure, frequency of occurrence, and backside thermal limitations. Spacecraft limitations regarding tolerance to thermal input and possible contaminants will be estimated. Design concepts and material selection will be primarily based on minimum weight; however, requirements imposed by different spacecraft may require alternative designs. A material testing program will be developed to evaluate the conceptual designs under a concentrated heat source, most probably concentrated sunlight. A major concern will be the possible need for vacuum testing.

## SOLAR DYNAMIC POWER SYSTEM

## TECHNOLOGY TASK 2-11

(Concluded)

7. BENEFITS: Protection of the receiver face (and other components, for that matter) from the concentrated sunlight is required, as in the normal course of operation, it will occasionally be necessary to defocus the concentrator.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Alternative approaches exist, ranging from possibly a tungsten shield backed with multi-layer insulation, to a material which wastes away and has a definite lifetime. Contamination from the latter may not be tolerable to some spacecraft. By contrast, the tungsten shield may be heavier.

9. TECHNICAL RISKS: Risks are minimized by recognizing that more than one design concept will probably be required to satisfy different applications. Proof of more than one design through testing provides alternative solutions.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 2-12

1. TECHNOLOGY TASK TITLE: Receiver Shell Material Testing (2.8.2)

2. MAJOR TECHNOLOGY GROUP TITLE: Receiver

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Receiver
   Other high-temperature components
   - Remote thermal storage
   - Etc.

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Select suitable materials for construction of the receiver shell, and which also may be useful in insulating other high-temperature components.

5. STATE-OF-THE-ART LEVEL: The higher cycle operating temperatures compared to Space Station CBC will require redesign of the shell composite thicknesses, and possibly selection of other materials. The proposed Space Station CBC design has not been fabricated or tested.

6. BRIEF WORK STATEMENT: Design, fabricate, and test shell wall segments to verify design durability and thermal loss (conduction). Design and fabricate a full-scale receiver in support of the heat pipe receiver test (Task 2-2) as appropriate to that task. Wall construction will include metal multi-layer foil insulation; therefore, it will be necessary to test at a high vacuum to forestall interlayer gas conduction and the possibility of high-temperature alteration of surface emissivity due to gas interaction. Design optimization of the receiver shell will concentrate on minimizing power system weight, which will be a tradeoff of shell weight and shell conduction losses.

## SOLAR DYNAMIC POWER SYSTEM

## TECHNOLOGY TASK 2-12

(Concluded)

7. BENEFITS: To achieve a light-weight, high-temperature composite wall with low heat loss for construction of the solar receiver shell, that may also be appropriate for insulation of other high-temperature power system components.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): A high-temperature receiver is necessary for the performance and weight improvements associated with use of LiF TES, in order to improve over the Space Station selected TES material.

9. TECHNICAL RISKS: Risks are minimized by testing several wall segment designs for durability and heat loss rate. Following design selection, it is recommended that the design be used to fabricate a receiver shell to support full-scale heat pipe receiver testing (Task 2-3).

## 11.5 POWER CONVERSION UNIT ADVANCED TECHNOLOGY TASKS

A total of seven ATP tasks were identified for the power conversion unit (PCU), as shown in figure 11.5-1. Descriptions of the tasks may be found at the end of section 11.5.

Of the seven ATP tasks recommended, six relate to the Stirling cycle and only one to the CBC. That one is upgrading the CBC to operate at temperatures associated with LiF TES. The balance of CBC issues are being addressed in the Space Station Program.

The technological advancement expected for the heat pipe Stirling engine is the basis for the power system performance advantages expected over the CBC engine. The recommendations presented for the Stirling engine coincide with the intent of the planned 25 kWe Space Stirling Engine (SSE) Program as presently understood. All of the recommended tasks should be incorporated in the SSE Program.

### 11.5.1 Stirling Engine Design And Analysis (Task 3-1)

These first four tasks are the essence of present solar Stirling engine needs, and should be a part of the planned NASA-LeRC sponsored 25 kWe SSE Program. The task recommends design of a 35-40 kWe solar Stirling engine at a ratio of $T_H/T_C$ ~2.7, which would essentially have the same thermal input rate as the SSE design at $T_H/T_C = 2$. The solar engine design would have to be capable of handling much higher thermal input rates (ranging from about 20% to 60% additional due to excess solar energy) in off-design operation at reduced $T_H/T_C$ ratio conditions, where drop-off in engine efficiency would be of virtually no importance. The task recommendations should be incorporated in the planned SSE program.

### 11.5.2 Design of Stirling Engine Heat Pipe Heater (Task 3-2)

The planned SSE program Stirling engine heater configuration is the heat pipe heater. The conceptual design work by Sunpower, MTI, and Rocketdyne must be reduced to engineering design, with adequate thermal, stress, engine

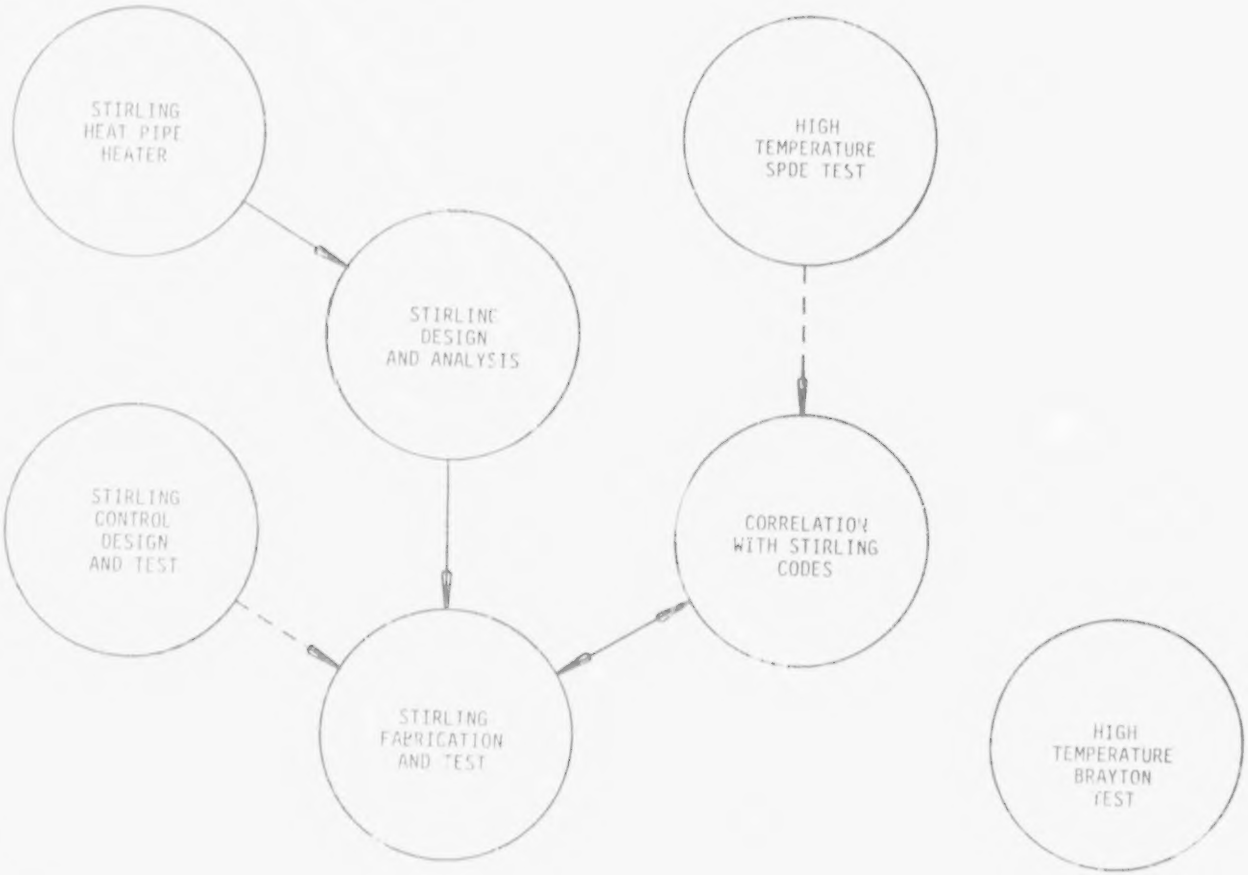

STIRLING HEAT PIPE HEATER
STIRLING DESIGN AND ANALYSIS
STIRLING CONTROL DESIGN AND TEST
STIRLING FABRICATION AND TEST
HIGH TEMPERATURE SPDE TEST
CORRELATION WITH STIRLING CODES
HIGH TEMPERATURE BRAYTON TEST

Figure 11.5.1. Power Conversion Unit Technology Tasks

performance, weight, and fabricability studies conducted for the heat pipe heater. This work requires collaborative system designer input regarding interfacing and interactions of the heater elements and the receiver/TES components. The task recommendations should be incorporated in the planned SSE program.

### 11.5.3 High-Temperature Stirling Engine (Task 3-3)

The SSE engine is presently planned to be operated with $T_H$ of about 1050K (1430F), which is a material-life imposed upper temperature limit, and is a value subject to possible further downward revision. The heat pipe receiver configuration (with LiF TES material) could be designed for somewhat higher $T_H$, perhaps 1075K± based on expected temperature drop in the heat pipe heaters; therefore, additional resistance will have to be built into the heat pipe/heater assembly to avoid exceeding 1050K. Recommended nominal design temperature for solar Stirling is 1033K corresponding to end-of-eclipse conditions (section 12) for the purpose of engine performance calculations. Operation of a FPSE at these temperatures will be a large step increase over previous Stirling programs. The task recommendations should be incorporated in the planned SSE program.

### 11.5.4 Stirling Engine Fabrication and Test (Task 3-4)

This is a customary and necessary step in the development process. It will be a prominent feature of the planned SSE program.

### 11.5.5 Stirling Engine Control Design and Testing (Task 3-5)

This task seeks to anticipate and test engine controls required for solar Stirling application. The baseline designs of this study incorporate the Space Station engine control scheme wherein all or part of the power may be dissipated with a parasitic load radiator (PLR). The PLR approach insulates the PCU from variation in load demand and allows for dissipation of excess energy as may be generated. A TES energy management control is also required such that the TES energy removed equals the energy stored in each orbit, so as to avoid over-temperature due to overcharging of the TES. This is a TES concern rather than an engine concern; however, the engine will no doubt be required to process the excess energy, as previously discussed. As a result, the solar Stirling engine control scheme shall have to be capable of controlling the engine operation up to perhaps 55-60% over nominal power. The task recommendations should be incorporated in the planned SSE Program.

### 11.5.6 Correlation of Stirling Engine Codes (Task 3-6)

The task recommends continuation of efforts to correlate FPSE design predictions and engine and/or component operating experience. The planned SSE program will increase both power and temperature over that of previous experience. The program also should provide data over a wider temperature ratio range for a given basic engine than previous experience, if the needs of both nuclear and solar Stirling are to be properly investigated. These codes are necessary to predict alternative design conditions and to predict off-design engine performance as well. The task recommendations should be incorporated in the planned SSE program.

### 11.5.7 High-Temperature Brayton Engine (Task 3-7)

This task recommends operation of a CBC power system at ~1100K turbine inlet temperature. Early in the Space Station design evolution, Garrett considered such a system to be within the present SOA. Since the CBC was chosen for Space Station solar dynamic power, the temperature upgrade would be a natural consideration for the Growth Station.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 3-1

1. TECHNOLOGY TASK TITLE: Stirling Engine Design and Analysis (3.3.2)

2. MAJOR TECHNOLOGY GROUP TITLE: PCU

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Stirling engine

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Determine by analysis engine design parameters and expected engine performance and specific weight at temperatures and temperature ratios associated with higher power solar dynamic systems. Provide sufficient parametric results to enable power system designers to simulate engine operation for design and off-design conditions.

5. STATE-OF-THE-ART LEVEL: Engine designs to date have been directed to nuclear heat sources, with temperature ratios normally $\leq 2.0$; whereas, solar may need ratios in the range of 2.2 to 3.0. Engine designs have not emphasized the importance of high efficiency at the expense of engine weight, as is needed for solar applications.

6. BRIEF WORK STATEMENT: Perform engine design and analysis for single cylinder FPSE at power levels of 35-40 kWe net power output, and designed for high efficiency and high temperature ratios appropriate to solar application (2.2 to 3.0 range). Parametric design data will be developed suitable for use by power system designers, including performance and specific weight versus temperature ratio for an engine hot temperature of ~1050K. The tradeoff of performance versus specific weight will be developed for at least temperature ratios of 2.4 and 2.8. Expressions for the computation of $T_H$ and $T_C$ will be determined for engines designed for both pumped heat transport loops and for heat pipe input to $T_H$ as

well. Nonmetallic fluids as well as the liquid metals will be considered for engine cooling. Design data relative to heater and cooler configurations will be developed, including general geometry and dimensions, to enable computation of interface conditions of flowrates, heat flow, pressure drop, etc. Consideration will be given to the eventual expansion of the analysis to other power levels and possibly higher temperatures.

7. BENEFITS: The benefits will be a more extensive parametric data base on Stirling engine design information, applicable to conditions appropriate to solar dynamic power systems, thereby improving the quality of power system trade studies.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): The present data base is quite inadequate for more than cursory power system trade studies. The absence of adequate parametric data will discourage system designers from considering the FPSE.

9. TECHNICAL RISKS: Technical risks lie in the adequacy of Stirling codes to be used in the parametric analysis. The codes will have to be validated by engine test, which will of necessity include only one or two design conditions. This process of test and validation must be a long-term project, wherein this is but a step along the way to successful engine development and code development.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 3-2

1. TECHNOLOGY TASK TITLE: Design of Stirling Engine Heat Pipe Heater (3.4.1)

2. MAJOR TECHNOLOGY GROUP TITLE: PCU

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Stirling engine
   Receiver

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Develop engine designs based on integration of the engine heaters and heat pipes for transport of heat to the engine. Collaborate with power system designers regarding integration of the heat pipes with the solar receiver and TES.

5. STATE-OF-THE-ART LEVEL: Sunpower has performed conceptual design work on the heat pipe heater. The only work which has been done integrating the heat pipe with the receiver and TES has been performed by Rocketdyne.

6. BRIEF WORK STATEMENT: Design and analyze integrated heat pipe heaters for the Stirling engine for a 35-40 kWe power level. Tradeoffs of maximum thermal power per heat pipe, integration of the engine heaters and coolers, fabricability, etc. will be performed. Performance improvements due to higher $T_H$ increasing $T_H/T_C$ (and therefore efficiency), engine weight changes, size of power system components due to higher engine efficiency, expected weight reduction in receiver/TES, etc. will also be examined. The effect upon heater design due to different engine power levels will be investigated. The work will be a cooperative effort between the Stirling engine designer and a power system designer due to the design interaction required for engine heat pipes, heat pipe receiver, and TES design. Design tradeoffs must ultimately be performed at the power system level, primarily to minimize power system weight.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 3-2
## (Concluded)

7. BENEFITS: Performance benefits expected of the heat pipe Stirling engine design are improved engine performance (efficiency), and reduction in overall power system weight. The passive nature of heat pipe heat transport eliminates the hot loop pump, and also allows for more graceful degradation should one or a few of the heat pipes fail.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Preliminary studies indicate that a pumped loop receiver power system would be heavier and less reliable.

9. TECHNICAL RISKS: Technical risks are minimized by continuing development of the pumped loop receiver as a backup design until the heat pipe engine/receiver approach is sufficiently developed to prove superior performance.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 3-3

1. TECHNOLOGY TASK TITLE: High-Temperature Stirling Engine (3.6.3)

2. MAJOR TECHNOLOGY GROUP TITLE: PCU

3. POWER SYSTEM COMPONENT(S) AFFECTED:

   Stirling engine

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Demonstrate successful operation and durability of a Stirling engine at 1050K $T_H$.

5. STATE-OF-THE-ART LEVEL: FPSE engines have not been operated at high temperature. The NASA-LeRC sponsored Space Power Demonstrator Engine (SPDE) operated at 630K $T_H$.

6. BRIEF WORK STATEMENT: Demonstrate reliable Stirling engine operation at ~1050K $T_H$. The 35-40 kWe Stirling engine will be designed for operation at ~1050K; however, through the course of development, early tests will be operated at derated conditions, including below nominal temperature. Successful extended duration operation at design conditions will ultimately be performed, proving the engine design and materials selections. Operation of existing engines at higher temperatures approaching ~1050K will also be investigated for feasibility, and a test program will be recommended, if appropriate.

7. BENEFITS: Operation at ~1050K is necessary to realize the performance benefits associated with higher Carnot efficiency and the LiF heat of fusion effect upon TES weight.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Lower performance, and a heavier power system, would be the result of using lower cycle inlet temperature.

## SOLAR DYNAMIC POWER SYSTEM

## TECHNOLOGY TASK 3-3

(Concluded)

9. TECHNICAL RISKS: Technical risks are minimized by the selection of engine temperature of ~1050K, as associated with LiF TES. The next higher TES salt is NaF, which would result in 140K - 160K higher engine inlet temperature, and although resulting in higher Carnot efficiency, does not offer any power system weight advantage over LiF TES. The ~1050K temperature corresponds to about a 45K temperature increase over the temperatures associated with the Space Station CBC cycle.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 3-4

1. TECHNOLOGY TASK TITLE: Stirling Engine Fabrication and Test (3.3.2)

2. MAJOR TECHNOLOGY GROUP TITLE: PCU

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Stirling engine

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Fabricate and test one or more higher power FPSE engines designed for solar dynamic power system operating conditions. Demonstrate achievement of engine design goals. Provide validation for Stirling codes.

5. STATE-OF-THE-ART LEVEL: The largest FPSE engine tested is the MTI 25 kWe demonstrator engine, with 12.5 kWe per cylinder. This engine has not achieved design goals, nor does it have self-acting dynamic hydrostatic gas bearings and other features desirable for space applications.

6. BRIEF WORK STATEMENT: Design, fabricate, and test a 35-40 kWe FPSE engine. Due to the absence of test experience at this power level and temperature, test planning will include the possibility of initially not achieving design goals, and that alternate design approaches may be necessary as is often the case with early development work. The test engines will be instrumented so as to provide adequate data to correlate engine operation with Stirling code predictions.

7. BENEFITS: The successful completion of testing at the prescribed power and temperature would allow the FPSE engine to be considered for space solar dynamic power systems, and for the apparent performance advantages of a Stirling power system to be proven as real.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 3-4
(Concluded)

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Without considerable test demonstration of FPSE engine performance and reliability, the engine cannot be considered for space use.

9. TECHNICAL RISKS: Well-planned test programs anticipate areas of uncertainty, perform component test where possible and provide extensive instrumentation in order to progress through the development in as orderly a fashion as possible. For the FPSE, initial testing will be at derated conditions of power and temperature, working up to design conditions stepwise as prior steps are accomplished successfully. Adequate funding will be required to achieve the development goals in a timely fashion.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 3-5

1. TECHNOLOGY TASK TITLE: Stirling Engine Control Design and Testing (3.8.4)

2. MAJOR TECHNOLOGY GROUP TITLE: PCU

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Stirling engine
   Receiver (possibly)
   Radiator (possibly)

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Demonstrate Stirling power system control for variations in input power and associated variation in $T_H$ and $T_C$ as would occur about typical orbits due to the range of solar energy input conditions.

5. STATE-OF-THE-ART LEVEL: Control technology for the FPSE engine conventionally controls power by adjustment of the gas charge in the engine, requiring an active subsystem for gas charge control, or by variation in power brought about changing applied alternator voltage.

6. BRIEF WORK STATEMENT: Design and demonstrate control systems for the FPSE power system for typical solar operation. Variations in $T_H$ and $T_C$ will be determined from analysis of typical missions. Control approaches will consider both engine control and schemes of energy management external to the engine. The primary evaluation criteria will be power system weight and control system reliability.

7. BENEFITS: The power system must be controlled in some way to cope with the variable energy environment experienced by space solar power systems. Early development and breadboard testing of control systems may, in turn, influence design of the engine and other components in order to achieve a minimum weight, reliable, controllable power system.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 3-5
## (Concluded)

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Development of a control system after the development of power system components may result in a less than optimum power system design; wasteful of energy, heavier, and possibly less reliable.

9. TECHNICAL RISKS: Technical risks are minimized by addressing the control system design as a separate and important task early in the design process. Breadboard testing of the control system will be performed before being incorporated with the power system components.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 3-6

1. TECHNOLOGY TASK TITLE: Correlation of Stirling Engine Codes (3.3.2)

2. MAJOR TECHNOLOGY GROUP TITLE: PCU

3. POWER SYSTEM COMPONENT(S) EFFECTED: Stirling engine

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Correlate Stirling code predictions with actual FPSE engine operation at design conditions (engine efficiency and weight), with code modification as necessary to obtain good correlation. Extend code capability to adequately predict engine operation for off-nominal operating conditions, including modelling of engine control systems.

5. STATE-OF-THE-ART LEVEL: A number of FPSE codes exist in industry, but as yet are more useful in indicating trends rather than being reliable in predicting actual engine operation. One reason for the uncertainty is limited data for code correlation, and an absence of experience at high power levels.

6. BRIEF WORK STATEMENT: Upgrade FPSE codes to more accurately represent Stirling engine operation (efficiency and specific mass). The several existing codes for FPSE engine characterization will be examined for suitability to predict engine design performance, and off-design operation. The correlation will utilize engine test results from past, present, and future engines to upgrade the codes. The upgraded codes will then be used to recommend design improvements, for future engine modification and testing. The code improvement task must be considered as open-ended, to be continued as long as FPSE engine design and development shall continue.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 3-6
(Concluded)

7. BENEFITS: It is normal practice in industry to create computer codes suitable for characterization of power conversion units (PCU), to enable engine designers to seek improvements in their product, and to enable system designers to conduct trade studies at the system level. Improved engine codes improve the quality of the system design in which the engine is used.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Present codes do not predict engine performance with sufficient accuracy by taking a given design and extrapolating to other design conditions. The more complicated models are not suited for parametric trade study use. Data being provided to system level designers is not of high confidence.

9. TECHNICAL RISKS: Availability of quality test results from various engine designs will depend on further testing of existing FPSE engines and testing of newly designed engines. This data may be slow in coming as development of new engines is a time-consuming endeavor. The important issue is that code development and engine development are each essential to the success of the other, and task planning and budgeting must recognize this issue.

C - 4

## SOLAR DYNAMIC POWER SYSTEM

## TECHNOLOGY TASK 3-7

1. TECHNOLOGY TASK TITLE: High-Temperature Brayton Engine (3.7.3)

2. MAJOR TECHNOLOGY GROUP TITLE: PCU

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Brayton engine

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Demonstrate successful operation and durability of a Brayton engine at ~1100K turbine inlet temperature.

5. STATE-OF-THE-ART LEVEL: The Brayton cycle for Space Station is to operate with a nominal ~1005K turbine inlet temperature. Components have been proven to ~1100K and higher, and Garrett originally proposed that Space Station CBC be operated near ~1100K turbine inlet temperature.

6. BRIEF WORK STATEMENT: Demonstrate reliable Brayton engine operation at ~1100K turbine inlet temperature or higher, as appropriate to a CBC design using LiF TES. The Space Station CBC design may eventually be upgraded in temperature for Growth Station. The engine will be fabricated and tested, first at derated conditions, and finally tested for extended duration at design conditions. The basic Space Station CBC engine is presently rated for about 32 kWe alternator output, which can be readily uprated to 35-40 kWe when redesigned for ~1100K.

7. BENEFITS: Operation at ~1100K is neces[illegible] to realize the performance benefits associated with higher Carnot efficiency and the LiF heat of fusion effect upon TES weight.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 3-7
## (Concluded)

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Lower performance, and a heavier power system, would be the result of using lower cycle inlet temperature.

9. TECHNICAL RISKS: Technical risks are minimized by the selection of turbine inlet temperature of ~1100K, as associated with LiF TES. The next higher TES salt is NaF, which would require a 140K higher turbine inlet temperature, and although resulting in higher Carnot efficiency, does not offer any significant power system weight advantage over LiF TES. The ~1100K temperature corresponds to a 80K temperature increase over the temperatures associated with the Space Station CBC cycle.

## 11.6 RADIATOR ADVANCED TECHNOLOGY TASKS

One ATP task was identified for the radiator, as described in Task 4-1. The titanium/methanol heat pipe radiator was chosen for application to both the CBC and Stirling power cycles. As described in section 10.6, the methanol upper operating temperature limitation presently restricts power system weight and area optimizations, and the restriction will be more pronounced with technological advancements in other subsystems. The technology task is directed at development primarily in two areas: the upper temperature limit for methanol must be better defined; and the water heat pipe must be further pursued to establish what materials, in particular titanium, are compatible with water. Each of these requires long term testing to be conducted, which should be commenced promptly as such knowledge is already needed for the conduct of systems trades. Examination of alternative heat pipe fluids and non-metallic materials of construction will be included.

## SOLAR DYNAMIC POWER SYSTEM

## TECHNOLOGY TASK 4-1

1. TECHNOLOGY TASK TITLE: Radiator Heat Pipe Selection (4.1.4)

2. MAJOR TECHNOLOGY GROUP TITLE: Radiator

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Radiator
   Stirling engine

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Develop a higher temperature heat pipe utilizing water or an alternate fluid, such as toluene, capable of operating up to 465K-475K (377F-395F). Trade studies indicate that higher radiator temperatures have the potential of reducing power system weight.

5. STATE-OF-THE-ART LEVEL: Water is thermally a fairly good heat pipe fluid, but has the disadvantages of a high freeze temperature and materials compatibility. Copper is the only heat pipe material known to be compatible with water. There is evidence that high-strength brass, or possibly one of the stainless steels could be used. It is not known whether titanium would be compatible with water.

6. BRIEF WORK STATEMENT: Expand upon current technology in water heat pipes and investigate the feasibility of alternative working fluids capable of operating up to 465K-475K. Establish the upper operating temperature limit for methanol. Laboratory scale extended-duration capsule testing will be used to expand upon the current knowledge of materials compatibility and thermal decomposition. Emphasis shall be placed on the use of titanium heat pipe material due to the weight savings potential. Alternate materials of construction shall be compared for heat pipe radiator application on the basis of minimum weight considering micrometeoroid and debris hazard.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 4-1
## (Concluded)

7. BENEFITS: The benefits sought will be lower power system weight and reduced radiator area for the solar dynamic power systems.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Solar dynamic engines would be operated at non-optimum temperature ratios. Both weight and area would be increased.

9. TECHNICAL RISKS: Alternative heat pipe designs may not be found to be successful in improving power system weight and area. Planning the development work stepwise, first with laboratory testing of materials and fluids, and finally with complete heat pipe testing, must be done so as to evaluate and require approval before proceeding with successive steps.

## 11.7 POWER SYSTEM ADVANCED TECHNOLOGY TASKS

Two ATP tasks were identified for the power system.

### 11.7.1 Excess Heat Rejection (Task 5-1)

A solar power system may be subject to as high as 60% excess energy input as a result of seasonal and orbit eclipse variations. Variable type orbits and/or variable duty cycle may also impose demands upon the power system to dispose of excess energy. Past trade study results have chosen to rely upon the engine and radiator to process the excess energy, thus effecting engine controls and the size of the engine and radiator. This task would reexamine the choices for excess energy management, with emphasis on the impact upon system reliability. The task would investigate the effect of excess energy management upon the generic solar dynamic power system design approach.

### 11.7.2 Computer Software Advancement (Task 5-2)

Development of a common solar dynamic power system code, including a component data base relating to various levels of SOA, is recommended as a joint or collaborative project between government and industry. Mission planners need information on performance of solar dynamic power systems for various time frames of application. As an emerging industry, the quality and realism of the information will be important in establishing credibility and saleability of solar dynamic power. As a joint effort, the proposed code would be readily acceptable to both private and public users. The code could be used to guide and promote the NASA ASD program.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 5-1

1. TECHNOLOGY TASK TITLE: Excess Heat Rejection (5.1.4)

2. MAJOR TECHNOLOGY GROUP TITLE: System

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   Concentrator (possibly)
   Receiver
   PCU
   Radiator (probably)

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Selection and test of recommended excess heat rejection designs with particular emphasis on developing designs for Stirling power systems.

5. STATE-OF-THE-ART LEVEL: Both the CBC and ORC cycle designs proposed for Space Station direct excess heat either through or around the engine to the waste heat radiator. In each case, this is facilitated by a fluid flow loop; whereas, passing heat to the radiator loop is more of a problem for Stirling.

6. BRIEF WORK STATEMENT: Perform tradeoffs, selection, and test of recommended heat rejection designs for both Brayton and Stirling power cycles, with emphasis on the latter. The relative proportion of heat which must be rejected due to seasonal and orbital variations will be determined from examination of typical missions. Alternative techniques will include avoidance of excess energy capture, heat rejection from the receiver/TES, and excess power production and dissipation. Heat rejection may be before the engine, or through or around the engine to the waste heat radiator. Tradeoffs will be performed at a power system level, considering control strategy, weight, area, and reliability. One or more recommended designs will be fabricated and breadboard tested.

### SOLAR DYNAMIC POWER SYSTEM
### TECHNOLOGY TASK 5-1
### (Concluded)

7. BENEFITS: Excess energy management must be considered early in the system design in order to arrive at an optimum design approach, since the design possibilities potentially impact any one of the major subsystems.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): Excess energy management cannot be ignored, and delaying dealing with the issue could easily lead to less than optimum solutions.

9. TECHNICAL RISKS: Technical risks are minimized by treating excess heat rejection as a separate task to be performed early in power system development work.

## SOLAR DYNAMIC POWER SYSTEM

## TECHNOLOGY TASK 5-2

1. TECHNOLOGY TASK TITLE: Computer Software Advancement (5.5.5)

2. MAJOR TECHNOLOGY GROUP TITLE: System

3. POWER SYSTEM COMPONENT(S) EFFECTED:

   All components

4. OBJECTIVE OF TECHNOLOGY ADVANCEMENT: Establish a code and data base development philosophy requiring formalization of power system level codes through preparation by work statement, approval cycles, delivery to NASA, and provision for periodic updating. Stirling engine code development is a separate task.

5. STATE-OF-THE-ART LEVEL: Power system code and data base development have been independently pursued by NASA and by contractors. Space Station power system work now provides designs with which to validate codes, and discrepancies of existing codes have been noted. The quality of power system trade studies is dependent on the quality of the codes used.

6. BRIEF WORK STATEMENT: Formalize power system code and data base development for the continued ranking of alternative power system designs, specifically, Brayton and Stirling. Choice of NASA versus outside contractor shall have to be made. The virtue of code development by an outside contractor is the industrial feedback to NASA, and this is the recommended approach. More NASA involvement is required in the power system code development: preparation of code to a statement of work, briefing and approval cycle, delivery of codes. documentation, and periodic code update. The periodic updating suggests an ongoing contract and a close working relationship with NASA due to the dynamic nature of the code and documentation.

## SOLAR DYNAMIC POWER SYSTEM
## TECHNOLOGY TASK 5-2
## (Concluded)

7. BENEFITS: Improvement in the quality of power system codes and Stirling engine codes will aid NASA substantially in directing the available resources to the most promising development areas. The codes will be available to NASA for in-house studies, NASA will be familiar with code capabilities and limitations, and the present dissimilarities in trade studies as performed by NASA and by contractors with their own codes would essentially be eliminated. The benefit of NASA having the code in-house is the timely assessment of the impact of new results received from A/D contracts as viewed from a power system standpoint.

8. IMPACT IF TECHNOLOGY NOT DEVELOPED (Performance, Reliability, Cost, Potential Alternatives): The present situation of separate NASA and contractor codes is somewhat a duplication of effort, but more importantly, is producing trade study results which are not in agreement.

9. TECHNICAL RISKS: The risks would be minimized by utilizing the work performed to date by NASA and by contractors (Rocketdyne, in particular). A collaborative effort could produce a quality product at the present level of code sophistication whereupon code upgrade areas would become apparent and additional code improvement work statements could be formulated.

## 12.0 TRANSIENT ANALYSIS OF THE HEAT PIPE STIRLING THERMAL INTERFACE

The work reported in this section corresponds to Task XI from the SOW.

A transient thermal analysis was conducted of the interface between the heat pipe heater head of the Stirling engine and the solar heat receiver/TES. The heat pipe/TES module interface design which resulted from the analysis is shown in figure 12.0-1. The primary heat pipe transports energy to the TES, and the outer secondary heat pipe transports that energy from the annular TES to the Stirling heat pipe heaters. The same cross-section dimensions were used for both the 35 kWe and 7 kWe Stirling engines; TES quantity was adjusted using different lengths and number of heat pipe/TES modules.

The analysis considered variations of diameters, TES material section thickness, and percent density of nickel felt added for thermal conductance enhancement. For the designs shown in figure 12.0-1, with 10% dense nickel felt, heat pipe wall temperature ($T_H$) was predicted to vary by 17K (31F) for the design minimum insolation orbit. The variation in $T_H$ was predicted to be only 0.5K (1F) difference between the two missions, although the 7 kWe mission orbit was 15 minutes longer duration. The 35 kWe mission (similar to Space Station) was examined for maximum insolation, with 22% higher available energy which resulted in $T_H$ variation of 20K (36F) although maximum $T_H$ was at all times lower than for the design orbit. The maximum insolation orbit for the 7 kWe mission (500 km, 60° inclination) is the condition of continuous sunlight; therefore, engine operation would be quasi-steady state.

Maximum Stirling engine $T_H$ was limited to 1050K (1430F), which is the design maximum for the 25 kWe SSE. Taking the 17K orbital variation from 1050K resulted in 1033K (1400F) nominal design temperature for minimum insolation conditions at sunrise with the TES completely depleted. A design variation with the primary heat pipe connected to the annular heat pipe space outside the TES, and the secondary heat pipe located in the center, had no effect upon the predicted variation in $T_H$.

Under the assumption that excess energy is to be processed through the engine, it is necessary to ensure that energy consumption plus losses equals

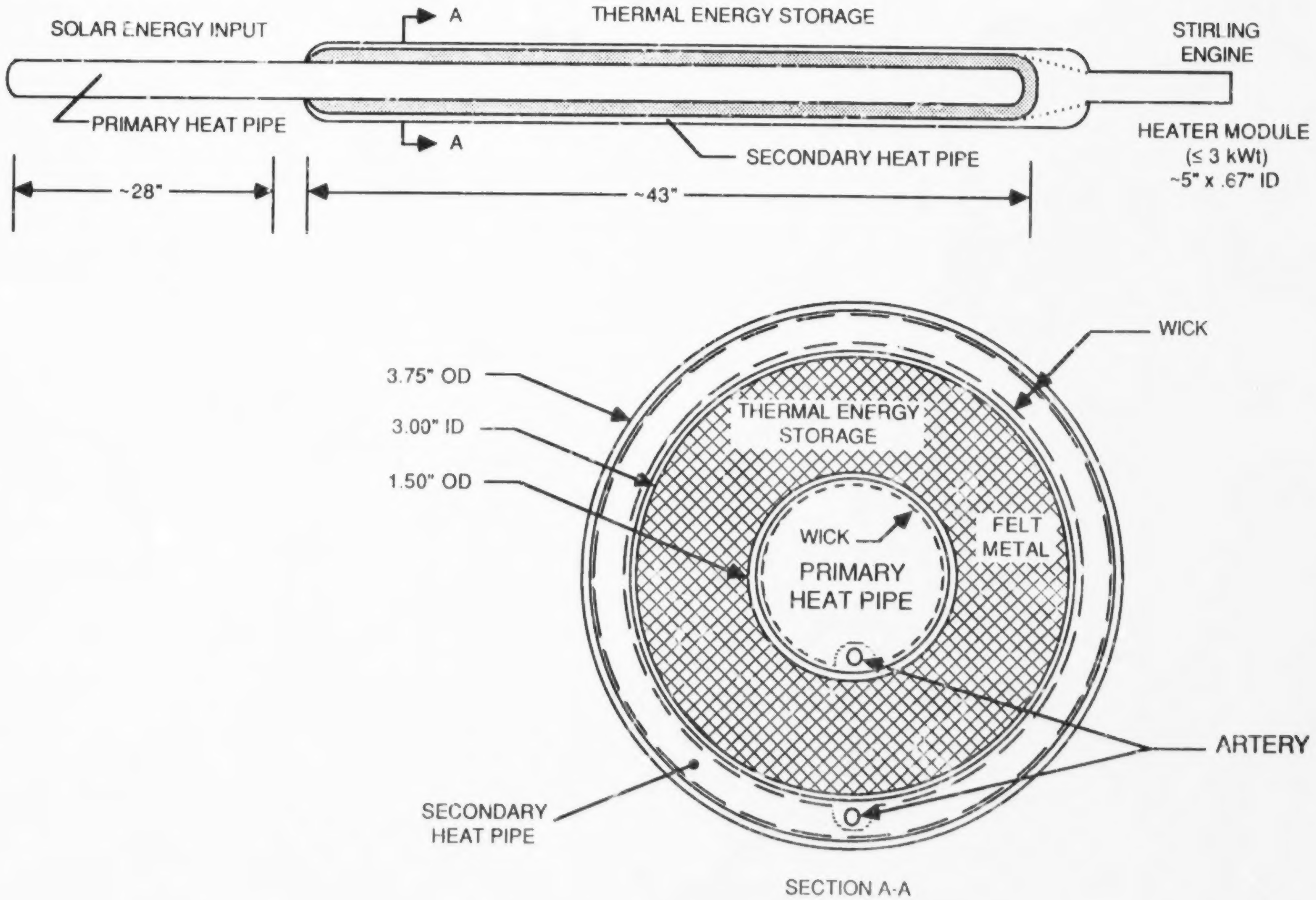

SOLAR ENERGY INPUT
A
THERMAL ENERGY STORAGE
STIRLING ENGINE
PRIMARY HEAT PIPE
A
SECONDARY HEAT PIPE
HEATER MODULE (≤ 3 kWt) ~5" x .67" ID
~28"
~43"
WICK
3.75" OD
3.00" ID
1.50" OD
THERMAL ENERGY STORAGE
WICK
PRIMARY HEAT PIPE
FELT METAL
ARTERY
SECONDARY HEAT PIPE
SECTION A-A

Figure 12.0-1. 35 kWe Stirling Engine Heat Pipe Energy Storage Module Design energy absorbed by the receiver for each orbit. The management aspect of the power system controller must ensure engine operation essentially at constant power output rate or a constant engine energy input rate such that the correct total amount of energy is consumed. For the transient analysis study, the assumption of constant engine energy input rate was made.

## 12.1 OPERATION OF THE SOLAR DYNAMIC POWER SYSTEM

Operation of the solar dynamic power system results in the condition of continuously varying thermal input/output conditions. The solar input changes due to eclipse and seasonal changes; and the TES is continuously being charged and discharged, affecting engine inlet temperature. Engine reject temperature changes as the radiator environment continuously changes and as the heat load changes with engine efficiency.

For this study, the power system was considered to operate in a base load mode, which is to provide constant power availability to the load. The Space Station design adds a requirement of providing capability for 115% engine power output for a fraction of an orbit, an added degree of variation in engine operating conditions. Both the Space Station design and this design include a parasitic load radiator to dissipate any unusable electric power.

Power system design requires analysis of these thermal variations, as they effect temperatures throughout the system, material stresses, and power system performance (power, efficiency, TES utilization, etc.). The Space Station CBC A/D receiver design by Boeing has been subjected to very extensive thermal transient analysis studies during the design evolution process.

The Stirling cycle differs from the CBC and Rankine cycles in that energy must be transported from the receiver/TES to the engine heaters, where the heat is transferred to the cycle working fluid sealed within the engine. The energy may be transported either by a pumped loop or by means of heat pipes. A design study of the 25 kWe SSE engine was conducted independently of this study, to determine the feasibility of a Stirling engine heater head which would employ heat pipes (the condenser section) in place of the original pumped loop heater head configuration. The results of the feasibility study were favorable, which opened up the possibility of a heat pipe solar receiver design such as depicted in figures 2.4-2 and 6.2.1.4-6, utilizing the heat pipe/TES module design shown in figure 12.0-1.

## 12.2 THERMAL TRANSIENT MODEL

The transient model considered solar energy absorbed by the receiver heat pipes, reradiation loss from the heat pipes through the receiver aperture, a TES insulation conduction loss, and a prescribed heat rate to the engine heaters. The small receiver conduction loss was lumped in with the TES insulation loss. The heat pipe modelling included wall and wick thermal capacity. The TES annulus was divided into ten equal-thickness layers. Separate values of heat capacity and thermal conductivity were used for the solid and liquid phases of the TES. Conductance enhancement of the liquid and solid phases was determined using the correlation by IGT (Institute of Gas Technology) developed for the Space Station CBC A/D receiver (L.M. Sedgwick, Boeing, Contract NAS3-24669):

$$k = (\text{percent}_{TES} * k_{TES}) + 0.56 * (\text{percent}_{FELT} * k_{FELT})$$

using percent by volume and k (thermal conductivity) for each material.

The transient code was set up to first size the heat pipe/TES modules (diameter, length, area, and weight), before setting up the necessary coefficients for the transient thermal analysis of the module. Inputs include: insolation level, eclipse and orbit interval, quantity of TES material, number of heat pipe/TES modules, felt metal percentage, various diameters and wall thicknesses, material properties, and various inputs to calculate thermal losses. Outputs include: heat pipe/TES module length and weight, input energy, output energy, losses, temperature throughout the circuit, and freeze-state of the TES material at each of the ten TES nodes (fraction frozen).

The thermal network circuit diagram is shown in figure 12.2-1 (symbols for the circuit diagram are defined in table 12.2-1). The network was solved with a finite element differential equation analyzer code developed by

Rockwell. The engine heater temperature $T_H$ was chosen as the heater gas-side wall temperature, T116.

Typically, cases were run for two complete orbits for given inputs and an assumed initial temperature distribution. Inputs and temperatures were then adjusted as necessary for another case run, and this procedure repeated several times until conditions of temperature and freeze-state were essentially duplicated at the start and end of the two-orbit run.

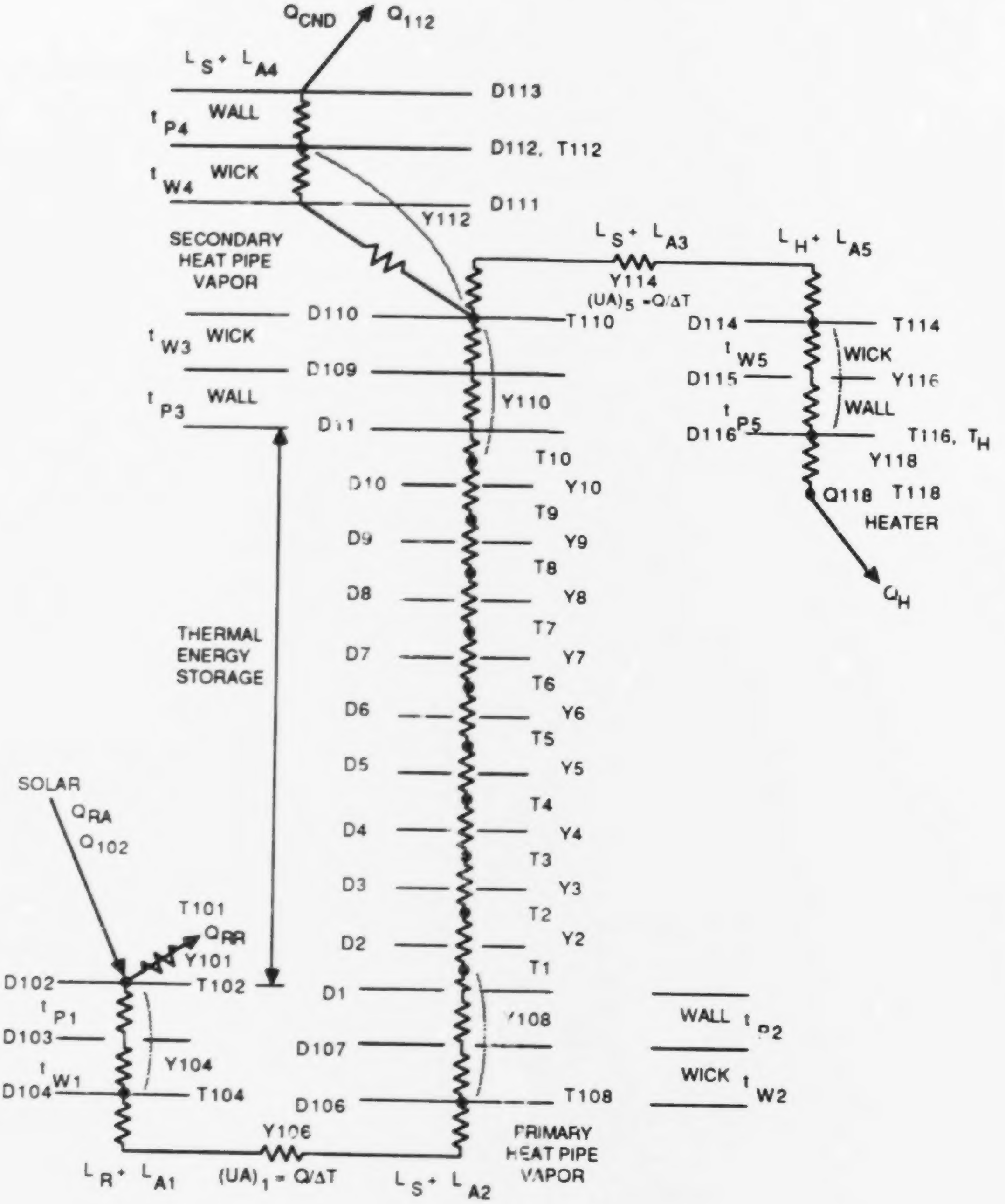
The following approximate orbit times were used for the transient analysis:

| Orbit |  | Orbit | Eclipse, min. |  |
|---|---|---|---|---|
| Altitude, km | Inclination | Period, min. | Maximum | Minimum |
| 500 | 28.5° | 95 | 35 | 27.5 |
| 1200 | 60° | 110 | 35 | 0 |

The differences between these times and the actual times would have an inconsequential effect upon the transient analysis results. Systems sizing results presented in section 6.4 used actual orbit times.

## 12.3 TRANSIENT THERMAL ANALYSIS RESULTS

A number of different heat pipe/TES module design cases were analyzed by varying geometry and felt metal density. Increases in TES section thickness reduced both length and weight; however, heater temperature variations about the orbit were greater. Increasing felt density increased weight and length, but reduced orbital temperature variations. The resulting engine heater temperature variations about the orbit are shown in figure 12.3-1. The lower curve is for a 4.45 cm (1.75 in.) outside diameter primary heat pipe and 1.59 cm (0.625 in.) annular TES thickness. The upper curve is for a 3.81 cm (1.50 in.) diameter heat pipe and 1.91 cm (0.750 in.) TES thickness. The higher single point is for a 3.81 cm (1.50 in.) heat pipe and 2.22 cm (0.875 in.) TES thickness. The lower single point is for a 3.81 cm (1.50 in.) heat pipe and 1.91 cm (0.750 in.) TES thickness, but with nickel-coated copper metal felt (40/60 Ni/Cu), with a much higher conductivity than the nickel felt.


$Q_{CND}$
$Q_{112}$
$L_S + L_{A4}$
D113
$t_{P4}$
WALL
D112, T112
$t_{W4}$
WICK
D111
Y112
SECONDARY HEAT PIPE VAPOR
$L_S + L_{A3}$
$L_H + L_{A5}$
Y114
$(UA)_5 = Q/\Delta T$
D110
T110
D114
T114
$t_{W3}$
WICK
$t_{W5}$
WICK
D109
D115
Y116
$t_{P3}$
WALL
Y110
$t_{P5}$
WALL
D11
D116
T116, $T_H$
Y118
Q118
T118
HEATER
$Q_H$
T10
D10
Y10
T9
D9
Y9
T8
D8
Y8
THERMAL ENERGY STORAGE
T7
D7
Y7
T6
D6
Y6
T5
D5
Y5
SOLAR
$Q_{RA}$
$Q_{102}$
T4
D4
Y4
T3
D3
Y3
T101
$Q_{RR}$
T2
D2
Y2
Y101
D102
T102
T1
D1
$t_{P1}$
Y108
WALL
$t_{P2}$
D103
D107
Y104
$t_{W1}$
WICK
$t_{W2}$
D104
T104
D106
T108
Y106
PRIMARY HEAT PIPE VAPOR
$L_R + L_{A1}$
$(UA)_1 = Q/\Delta T$
$L_S + L_{A2}$

NOTE: See Table 12.2-1 for list of symbols

Figure 12.2-1. Stirling Engine Heat Pipe Energy Storage Thermal Model

Table 12.2-1. Symbols for the Stirling Engine Heat Pipe Energy Storage Thermal Model

## SYMBOLS

| | |
|---|---|
| A | area |
| D | diameter |
| L | length |
| Q | heat transfer rate |
| T | temperature |
| t | thickness |
| U | overall heat transfer coefficient |
| Y | thermal admittance |

## SUBSCRIPTS

| | |
|---|---|
| A | adiabatic section |
| CND | conduction |
| H | engine heater |
| P | heat pipe wall |
| R | solar receiver |
| RA | radiation |
| RR | reradiation |
| S | thermal energy storage material |
| W | heat pipe wick |

Table 12.3-1 presents size and weight information for several selected design configurations, including the 35 kWe and 7 kWe design cases. The chosen nominal design cases with 16% dense nickel felt metal result in approximately a 17K (31F) variation in engine heater temperature for both the 35 kWe and 7 kWe mission orbits.

The orbital temperature variation of the receiver primary heat pipe wall temperature and of the engine heater gas-side wall temperature are shown on figures 12.3-2 through 12.3-5. The computed temperature traces are shown for the period of two orbits, beginning at sunrise. Figure 12.3-2 presents the results of the 35 kWe maximum eclipse design case (case number 9, see table 12.3-1) and figure 12.3-3 presents results of the same design for minimum eclipse. Figure 12.3-4 presents the results for the case of 15% dense 40/60 Ni/Cu felt (case number 5), which results in lower receiver heat pipe temperatures and reduced orbital temperature variations. Figure 12.3-5 presents the results of the 7 kWe maximum eclipse design case (case number 11).

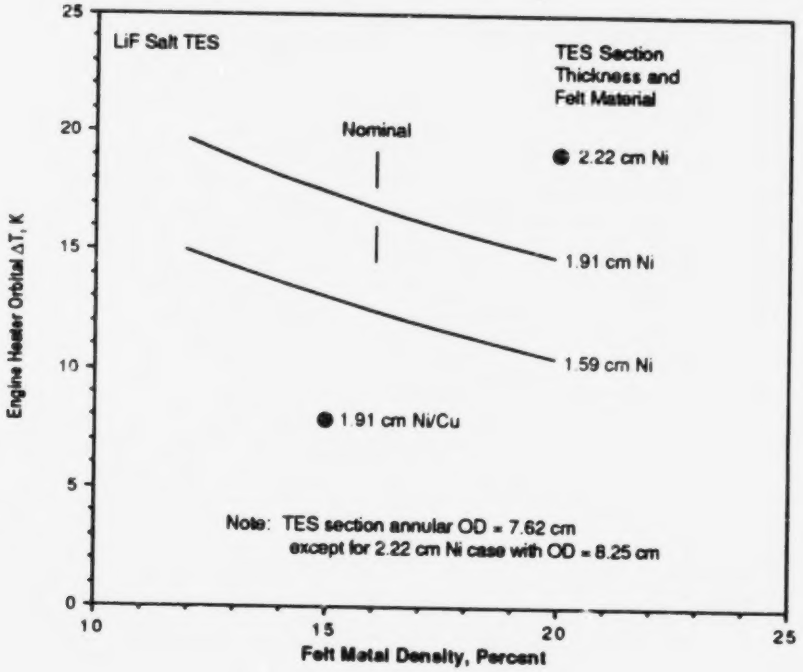


Figure 12.3-1. Stirling Engine Orbital Temperature Variation

Table 12.3-1. Heat Pipe/TES Unit Design Parameters

| Case number | 3 | 9 | 7 | 5 | 11 |
|---|---|---|---|---|---|
| Design power, kWe | 35 | 35(1) | 35 | 35 | 7(1) |
| Orbit time, min. | 95 | 95 | 95 | 95 | 10 |
| Eclipse time, min. | 35 | 35 | 35 | 35 | 35 |
| LiF weight, kg | 220 | 220 | 220 | 220 | 47.2 |
| Number of modules | 40 | 40 | 40 | 40 | 16 |
| Felt density, % | 12 | 16 | 20 | 15 | 16 |
| Felt material | Ni | Ni | Ni | Ni/Cu | Ni |
| Primary heat pipe OD, cm | 4.45 | 3.81 | 3.81 | 3.81 | 3.81 |
| LiF thickness, cm | 1.59 | 1.91 | 2.22 | 1.91 | 1.91 |
| Module length, m | 1.18 | 1.09 | 0.93 | 1.09 | 0.58 |
| Total module weight, kg(2) | 851 | 862 | 890 | 850 | 187 |
| Orbital ΔT variation, K | 14.8 | 16.7 | 18.9 | 7.7 | 17.2 |

Notes: 1. Nominal design cases.

2. Includes TES material, felt metal, primary heat pipe condenser and secondary heat pipe evaporator sections; excludes primary heat pipe evaporator and secondary heat pipe condenser sections.

Stirling Heat Pipe Receiver/TES Model - 35 kWe (1 of 40 Modules)
Case No. 9 - Maximum Eclipse 35 min./Orbit 95 min.
16% Felt Density, 1.50 in. Heat Pipe OD, 0.75 in. Salt Thickness

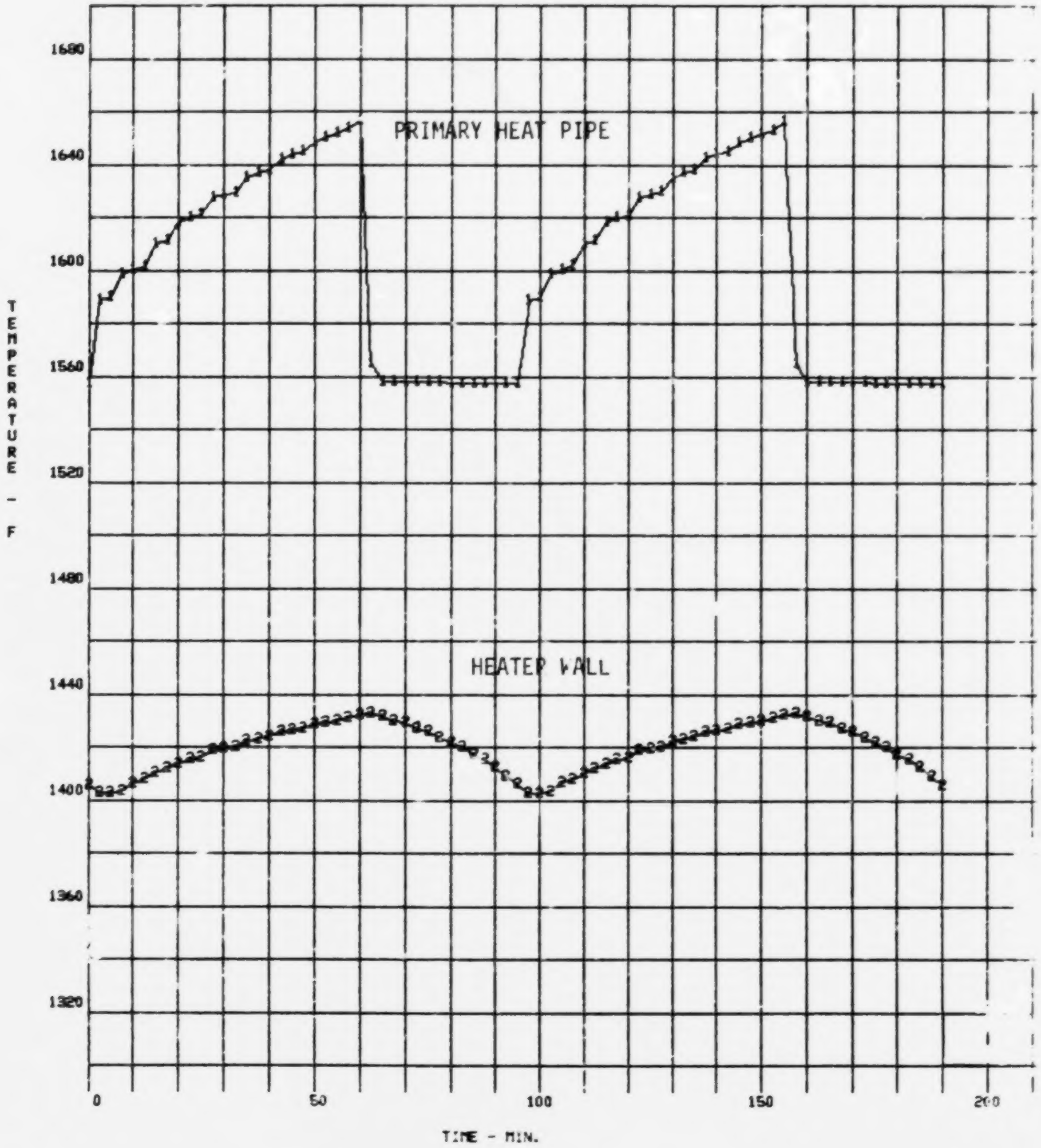


Figure 12.3-2. Heat Pipe Temperatures - 35 kWe Nominal Design

Stirling Heat Pipe Receiver/TES Model - 35 kWe (1 of 40 Modules)

Case No. 10 - Minimum Eclipse 27.5 min./Orbit 95 min.

16% Felt Density, 1.50 in. Heat Pipe OD, 0.75 in. Salt Thickness

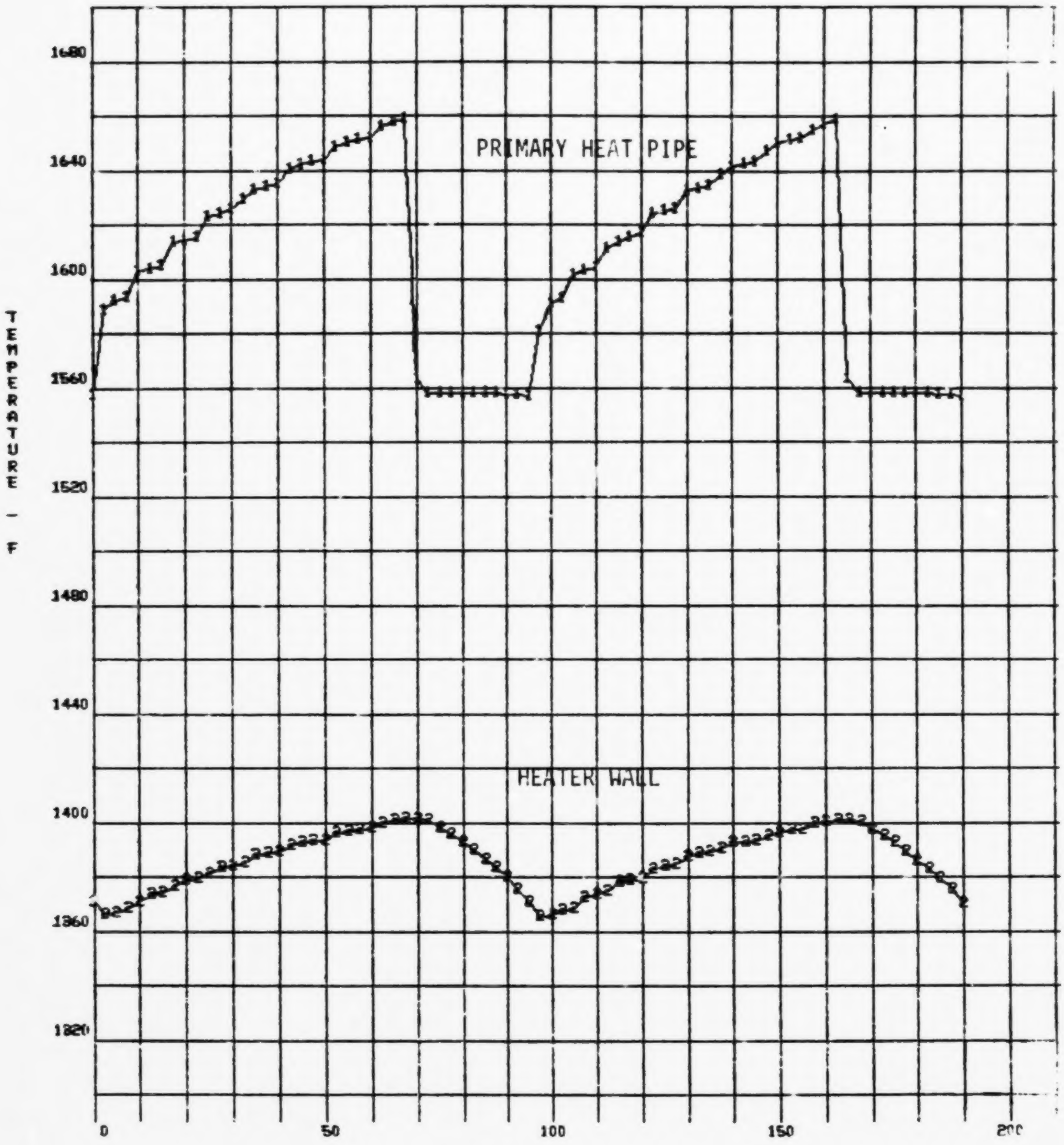


Figure 12.3-3. Heat Pipe Temperatures - 35 kWe Minimum Eclipse

Stirling Heat Pipe Receiver/TES Model - 35 kWe (1 of 40 Modules)

Case No. 5 - Maximum Eclipse 35 min./Orbit 95 min.

15% Ni/Cu Felt Density, 1.50 in. Heat Pipe OD, 0.75 in. Salt Thickness

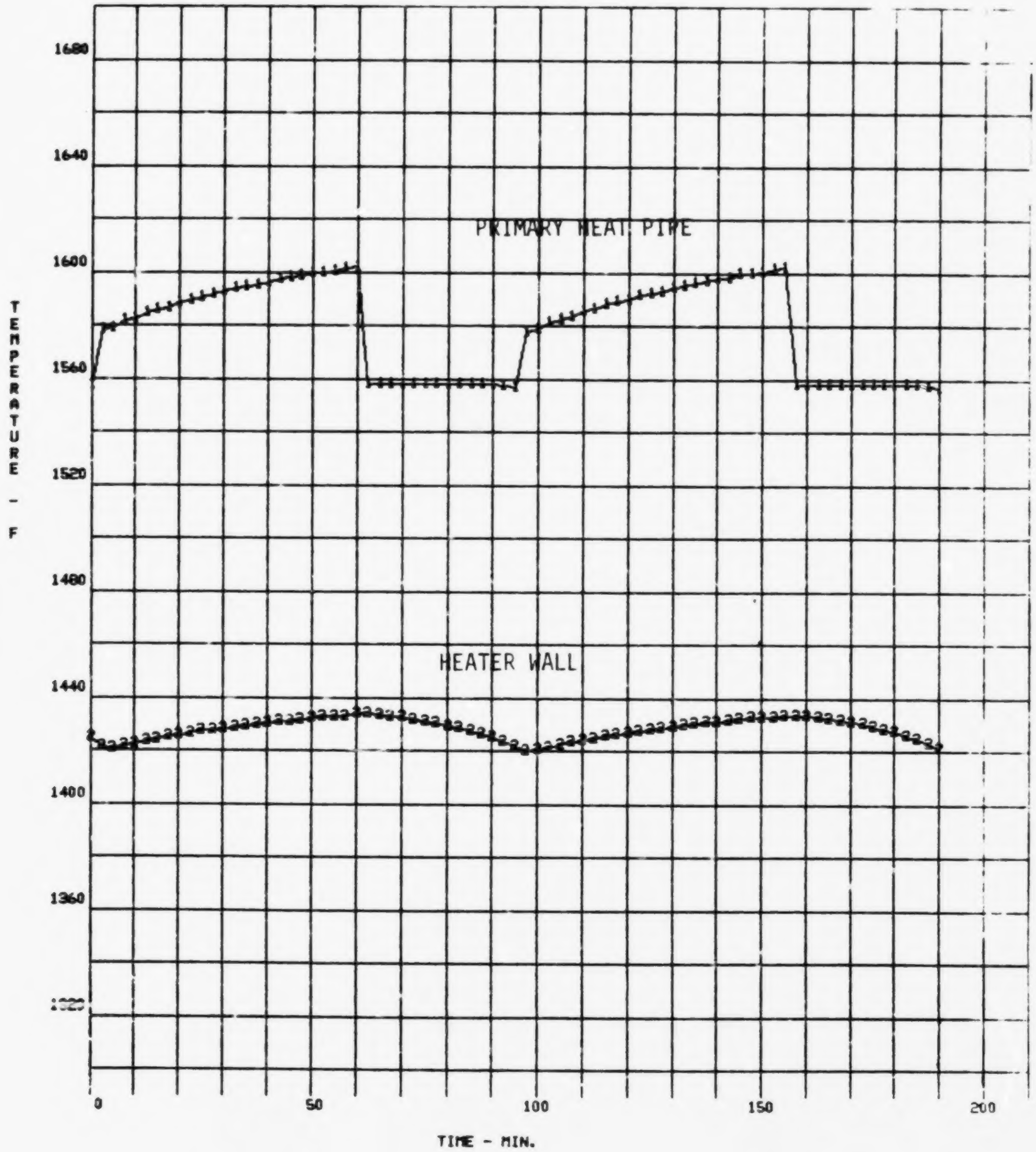


Figure 12.3-4. Heat Pipe Temperatures - 35 kWe Ni/Cu Felt

Stirling Heat Pipe Receiver/TES Model - 7 kWe (1 of 16 Modules)

Case No. 11 - Maximum Eclipse 35 min./Orbit 110 min.

16% Felt Density, 1.50 in. Heat Pipe OD, 0.75 in. Salt Thickness

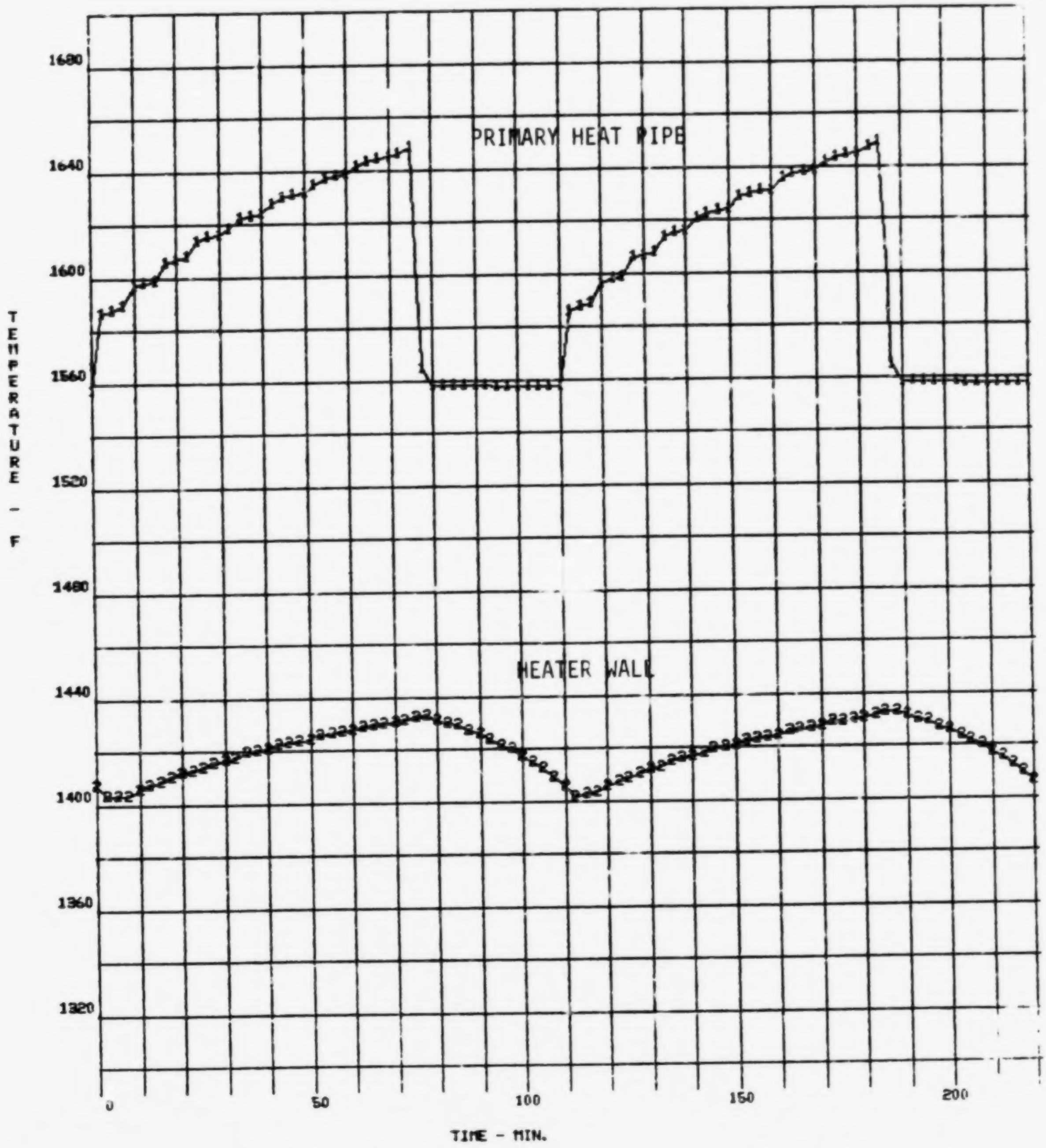


Figure 12.3-5. Heat Pipe Temperatures - 7 kWe Nominal Design

# 13.0 SOLAR DYNAMIC POWER SYSTEM HARDENING SUMMARY

The work reported in this section corresponds to Tasks VII and VIII from the SOW. Detailed results have been reported separately.

## 13.1 OBJECTIVE

Solar dynamic power systems could be considered as power sources on military space satellites, provided that their potential for survival in a hostile environment is defined. This effort represents a first step in defining the survivability potential of solar dynamic power systems.

The first objective of this effort was to provide a preliminary definition of the survivability level of the solar dynamic power system designs that evolved from the system definition studies. The second objective was to determine the level to which the designs could be hardened and to estimate the mass and/or performance penalties what would be incurred as a result of such hardening. The detailed results of this study activity are classified and are reported under separate cover (ref. 6). This section contains an unclassified summary of the system hardening study.

## 13.2 SUMMARY OF RESULTS

The purpose of the original SDPSD study was comparison and selection of power system design concepts for future NASA, civil, and military missions; however, survivability to other than natural threats was not a factor in the design selection process. The intent of the survivability study was to use the conceptual design as a starting point; first to determine survivability potential as designed, and second to determine how the designs might be made more survivable.

The analysis indicates that solar dynamic power systems can withstand all natural threats associated with earth orbital operation except for operation within the Van Allen Belt; however, unless specific design provision is made for hostile threats, the systems will have low resistance to the hostile threats. The study did indicate that substantial resistance to hostile threats can be built into a solar dynamic power system. It is important to note that specific mission requirements must be known during the early stages of the design process.

Resistance to nuclear and laser weapons could be improved by approximately one order of magnitude. Antisatellite pellets can be accommodated by armoring, and a design to accommodate one hit by a large kinetic energy weapon (KEW) appears to be feasible through the use of a redundant system concepts. Hardening the system to these levels would result in a mass penalty for the system of approximately 70%.

A brief study of advanced (high temperature) systems indicated that extremely high levels of nuclear weapon and laser threat could be accommodated by careful design of the exposed components. However, it did not appear likely that substantially greater resistance to neutral particle beam (NPB) or KEW attack would be possible. These threats are best accommodated by the use of maneuverability to avoid attack and the use of shoot-back to prevent attack.

General conclusions drawn from the hardening study are as follows:

- Current solar dynamic system designs are not totally resistant to the hostile environment envisioned by military needs
- Current designs can be hardened by approximately one order of magnitude via materials and configuration changes
- Advanced high-temperature design concepts can improve hardness by several orders of magnitude
- Substantial mass penalties are associated with hardening current designs

Solar dynamic systems do have substantial survivability potential provided they are designed for hostile threat resistance. The results are itemized in table 13.2-1.

Whereas this study was perhaps the first to ascertain solar dynamic power system hardening potential, a number of recommendations relative to further effort are appropriate. These are listed in table 13.2-2. The primary

Table 13.2-1. Hardening Study Quantitative Conclusions

---

- Substantial threat resistance can be designed into solar dynamic power systems
  - JCS = 12.0 study baseline
  - SMATH III
  - Accommodate pellet threat
  - Single missile hit
  - Up to 30 seconds of NPB threat
  - Maneuver to avoid missiles and NPB
- Advanced concepts could provide significant JCS and SMATH improvements
- Missiles and NPBs are most difficult threats to accommodate

---

Table 13.2-2. Hardening Study Recommendations

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- Detail study of survivability potential of a single specific design
  - 7 kWe Stirling, specified orbit
- Develop advanced materials
  - Beryllium mirror for concentrators
  - Advanced armors
  - Carbon/carbon composites
  - Optical filters
- Investigate alternative concepts for impact resistance
- Investigate alternative concepts for NPB resistance
- Conduct small-scale tests to verify hardening concepts
  - Laser irradiation tests
  - Flash x-ray test (in conjunction with other underground nuclear tests)
  - Impact tests

---

recommendation is a detailed design study for a specific mission or missions. Such a study would help definitize mass penalties for hardening of the power systems. In parallel with this effort, advanced development activities to develop hardened components are also recommended. These combined with small scale laboratory testing for preliminary verification would help bring hardened solar dynamic systems to fruition.

## 14.0 CONCLUDING REMARKS

The Solar Dynamic Power System Definition Study has shown that solar dynamic power offers the potential of significantly reduced power system weight and area for low- and mid-earth orbits as compared to photovoltaic power, based on near-term state-of-the-art. A somewhat smaller advantage would be predicted for a geosynchronous orbit. The study found that future growth in power level requirements will see much higher power levels needed for the low altitude orbits than for geosynchronous orbits. The study also confirmed that scaling of solar dynamic power systems from lower to higher power results in significant reductions in specific weight (kg/kWe), more so than for scaling of photovoltaic power systems. The combined growth in power requirements and improvement in specific weight are the major factors which produce the significant performance advantages of solar dynamic power as indicated for the power range of this study (7 kWe and 35 kWe) and higher.

For solar power production in a hostile threat environment, results of this study indicate that solar dynamic power systems may be hardened substantially against a variety of threats, with increases in system weight.

The study shows two distinct areas of performance improvement for solar dynamic power, and these serve as major issues in the recommended advanced technology program. First is utilization of LiF salt for thermal energy storage, having a heat of fusion 32% greater than the $LiF-CaF_2$ eutectic to be utilized for Space Station. (This improvement is partially offset by 16% higher liquid specific volume). A higher fraction of Carnot efficiency is also realized due to LiF melting temperature being 79K (142F) higher than the eutectic. The higher temperatures will challenge superalloy materials limitations severely, and may require use of refractory alloys in regions of highest temperatures, such as the receiver. The conceptual designs for this study attempted to avoid the need to use refractory alloys in the engines, for both Brayton and Stirling cycles.

The second area of performance improvement is that of the Stirling cycle as compared to the Brayton cycle. Power system weight and area reduction of approximately 20% are predicted for the heat pipe receiver Stirling cycle configuration. NASA-LeRC is embarking on a 5-year development program for the 25 kWe Space Stirling Engine (SSE), designed for a nuclear application with a temperature ratio of 2.0. For solar application, optimum temperature ratio is more in the range of 2.6-3.0 (with a fixed engine hot temperature of ≤1050K (1430F)). The higher temperature ratios result in much higher Carnot efficiencies and power output; for the same thermal input, the 25 kWe SSE would be capable of 34-39 kWe power output. Operation of the engine cold end at temperatures between about 350-400K (170-260F) greatly relieves the alternator materials and cooling concerns as compared to the 525K (485F) nominal cold end temperature for the 25 kWe SSE.

The 25 kWe SSE program does not at present include a solar engine design version. This study recommends that the solar design be added to the SSE program, if only in the form of a paper design, so as to ascertain the basic differences as may arise between the two applications, nuclear and solar.

Another task recommendation from this study is creation of a standard computer software code and data base for solar dynamic power, using a collaborative activity between government and private industry. The objective would be a code readily acceptable to both private and public users, thereby diminishing disparities between the public and private sectors regarding power system performance as currently exists in published literature. The customers, the mission planners, need the reliable input possible with a standard solar dynamic power system code and data base. A code such as this could be the centerpiece in guiding the NASA Advanced Solar Dynamic Program.

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16. Abstract

The solar dynamic power system design and analysis study compared Brayton, alkali-metal Rankine, and free piston Stirling cycles with silicon planar and GaAs concentrator photovoltaic power systems for application to missions beyond the Phase II Space Station using Space Station level of technology for all power systems. Conceptual designs for Brayton and Stirling power systems were developed for 35 kWe and 7 kWe power levels. All power systems were designed for 7-year end-of-life conditions in low earth orbit. LiF was selected for thermal energy storage for the solar dynamic systems. The study results indicate that the Stirling cycle systems have the highest performance (lowest weight and area) followed by the Brayton cycle, with the photovoltaic systems considerably lower in performance. For example, based on the performance assumptions used in the study, the planar silicon power system weight was 55-75% higher than for the Stirling system.

A technology program was developed to address areas wherein significant performance improvements could be realized relative to the current state-of-the-art as represented by Space Station. In addition, a preliminary evaluation of hardenability potential found that solar dynamic systems can be hardened well beyond the hardness inherent in the conceptual designs of this study.

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